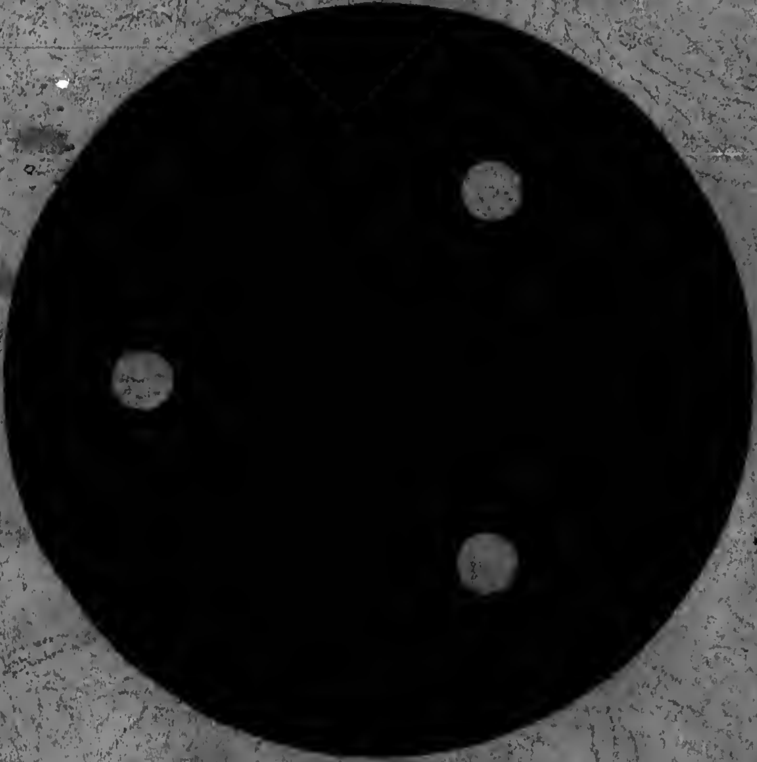
PROCEEDINGS
# of the
FIRST SPACE CONGRESS

APRIL 20, 21 & 22, 1964
COCOA BEACH, FLORIDA



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FIRST SPACE CONGRESS

PRESENTS

"WHERE ARE WE GOING IN SPACE?"

APRIL 20, 21, 22, 1964

Cocoa Beach, Florida

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Canaveral Council of Technical Societies

JOHN T. HAGAN
General Chairman

CHAIRMAN'S MESSAGE

The First Space Congress has been a joint endeavor of nineteen professional societies and two colleges which comprise the Canaveral Council of Technical Societies. Starting in 1959 with the goal of better communications among the many technical societies, the Council has grown to become a leader in our community. The Proceedings of this First Space Congress represent our most significant undertaking to date. It confirms the Councils' original thinking that many diverse technical professions with common interests and objectives can effectively coordinate activities to the benefit of all. It is hoped that our example will be followed in other parts of the country for the further benefit of other technical societies and the many organizations which support their activities.

The specific purpose of the First Space Congress has been to explore our theme, "Where Are We Going In Space?" The papers contained herein give many answers to this question from different professional points of view. Collectively, they emphasize the progress being made in present programs and technologies and look forward to the complexities of the continuing challenge that we face. We have been very fortunate in obtaining such a diversity of exceptionally qualified authors. We are indeed grateful to them, and also for the sincere interest shown by our distinguished principal speakers.

The Canaveral Council of Technical Societies is justly proud to publish these Proceedings of our First Space Congress, and sincerely hopes that they will be of continuing benefit to future readers. The Council intends this to be only the first of an annual space congress to be held here in the heart of the free world's space activities.

John T. Hagan
General Chairman
Space Congress Committee

FIRST SPACE CONGRESS - 1964

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THE PROGRAM

THE CONGRESS this year is aimed alternately at a number of specific fields in the necessarily broad interest area of space technology. Our motives for the broad coverage were twofold. First, we wished to make the program as attractive as possible to our wide range of contributing technical societies. Second, to begin to measure where we are going in space we had to sample the progress at many points along the broad frontier of our technology. Each succeeding year of this event will provide a more effective means of communicating the progress of the work which intimately affects our advancement in the Space Age. We hope we have added a sound block to the foundation of this endeavor.

Our highest expectations for speaker response were met in full measure. The unfortunate consequence of receiving so many outstanding papers is the necessity of having to turn some down for the lack of program time. To these and all the authors, guest speakers, Program Committee Members and advisors I wish to express my thanks for a valuable and successful First Space Congress.

William J. Haberhern
Program Chairman

SPEAKERS

APRIL 20, 1964

MR. JOHN T. HAGAN	*Introduction and Welcome*
DR. WALTER C. WILLIAMS	*Opening Address*
HON. EDWARD J. GURNEY, U. S. Representative, 11th Dist.	*Luncheon Speaker*

APRIL 21, 1964

MAJ. GEN. JOHN. B. MEDARIS, (USA-Ret)	*Luncheon Speaker*
GEN. BERNARD A. SCHRIEVER, USAF, Commander, AF Systems Command	*Keynote Address*

APRIL 22, 1964

MAJ. GEN. WAYMOND A. DAVIS, USAF, Commander AF Ballistic Systems Command	*Luncheon Speaker*

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NOTATIONS

INDUSTRY PARTICIPATION

The Congress will feature technical exhibits encompassing specialized products of the Space Industry. Exhibits will be located in the Ramada Inn and in the Gemini Room of Schrafft's Carriage House. They will be open from 0800 to 2000 during the Congress.

The following companies will exhibit .

Brandons, Inc., Jacksonville, Fla.

Cape, Inc., Melbourne, Fla.

Color Tran Industries, Burbank, Cal.

Eastman Kodak Co., Rochester, N. Y.

Ehrenreich Photo Optical, Inc., Palo Alto, Cal.

General Electric Co., Cocoa Beach, Fla.

General Analine & Film Corp., Binghamton, N. Y.

International Tel. & Tel. Corp., New York, N. Y.

Kern Instruments Inc., Port Chester, N. Y.

Keuffel & Esser, Hoboken, N. J.

Lee Associates, Inc., Orlando, Fla.

3M Company-Revere-Wollensak Div., St. Paul, Minn.

McDonnell Aircraft Corp., St. Louis, Mo.

NASA Public Information Office , Cocoa Beach, Fla.

NASA- Space Mobile Service Branch, Washington, D. C.

Phelps Dodge Electronic Products Corp., North Haven, Conn.

Traid Corporation, Encino, Cal.

U.S.A.F. Bio-Astronautical Exhibit, PAFB, Fla.

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OPTICAL COMMUNICATION SYSTEM ACQUISITION AND TRACKING

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Sylvania Electric Products Inc.
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Summary

The problem of acquiring and tracking a distant terminal for long-range laser communications is defined and solutions suggested. Factors considered include the effects of the atmosphere, the transit time, and the inherent limitations of optical systems.

Various parametric tradeoffs can be made in the synthesis of a system capable of the required accuracies. Some of the parameters have basic, or inherent, limitations while others are limited by size, weight, or cost considerations. These parameters are defined and their magnitudes established.

Finally, various system components are examined and their necessary characteristics defined to provide guidelines for the future development of equipments capable of performing the required tasks.

Introduction

The advantages to be gained from the inherent broadband capabilities of the laser in long-range communications require solutions to the acquisition and tracking problems associated with the use of extremely narrow beams. In order to evolve acquisition and tracking logic, it is necessary to examine in detail the various factors which govern system characteristics and to choose a transmitting beamwidth on the basis of available laser power, detector sensitivity, area of receiving aperture, and atmospheric perturbations (if transmission through the earth's atmosphere is to be considered). Knowledge of the magnitude of these parameters together with coarse acquisition information is then used to formulate acquisition systems.

Three acquisition systems are considered in this paper: one which holds the transmitter and receiver beamwidths constant throughout the acquisition cycle, one which holds the transmitter beamwidth constant but varies the instantaneous field of view of the receiver, and one which uses a storage device with memory. Common to all systems is the problem of external noise caused by the stars, earth and planets. These noise sources complicate the acquisition problem by adding the requirement of target discrimination.

The tracking problem is characterized by the high degree of precision necessary and the generally slow angular scanning rates. Ideally, the error-signal generation portion of the system should be compatible with the acquisition scanner to reduce interface and switch-over problems. A difficulty encountered in tracking extremely narrow beams with moving systems lies in the requirement that the transmitter "lead" the target to compensate for the error caused by the time it takes for the beam to travel the distance between transmitter and receiver. Additional complications arise when transmission through the atmosphere is necessary.

It is the purpose of this paper to examine the spectrum of system considerations outlined above which impact upon the total acquisition and tracking problem. As developed in the paper, the successful solution of the acquisition and tracking problem requires extreme precision in the various components which comprise the system. Most of the components can be made available now but others, such as precision beam deflectors, require further development.

The Acquisition Problem

The problem of acquisition is one of locating and illuminating a target with a narrow beam in a field of view which may be larger than the width of the beam. The size of the field of view is governed by the accuracy to which the position of the target is known.

The narrow beam associated with a system of this type makes the laser unsuitable for a general search over large volumes. For example, a laser beam with an angular width of 5 seconds by 5 seconds requires 1.3×10^{10} beam positions to search out a hemisphere. A beam position here is defined as the solid angle subtended by the half power points of the beam. If a dwell time of one microsecond is allowed at each beam position, a total time of 3.6 hours is required to complete the search. Because the relative motion between a space vehicle and an earth station does not permit this length of time to establish contact, it is necessary to perform coarse acquisition by some means external to the optical system.

If the acquisition is to be performed for a ground terminal, one of the precision radars can be used for coarse acquisition. The following list shows the angle accuracies of some of the radars presently in use. After a space vehicle has been tracked for several days, its position can be computed to much greater accuracies, certainly less than 1 second of arc.

Radar	Angular Accuracy	Frequency Band
AN/FPS-16	0.012°	C
Modified AN/FPS-16	0.0084°	C
AN/FPQ-6	0.006°	C
Tradex	0.017°	UHF

Since the space vehicle will not have the benefit of a high resolution radar, it will have to rely on an attitude stabilization system or far body tracker. The following list shows the presently attainable accuracies of several basic attitude stabilization systems.

Gravity Stabilization without Damping	±6°
Gravity Stabilization with Damping	±1°
Spin Stabilization	±1°
Horizon or Area Scanning	±1° or less
Sun, Moon, or Star Tracking	±1' to ±1°

It is also useful to examine the attitude control requirements for the OAO satellite. Not all of the accuracies shown in the following list have been achieved in flight but all are felt to be within or near the state of the art.

Initial Orientation	
Align toward Sun	0.5° to 1°
Acquire Star	0.5°
Acquire Earth	several degrees
Attitude Stabilization	
Toward Sun	1 second to several degrees
Toward Star	0.1 second
Toward Vertical	0.1° to several degrees
Zero Yaw Angle	1° to several degrees

These, then, represent the magnitude of coarse acquisition information which can be available for use on a laser based system.

A situation may occur in which a space vehicle looking toward the earth is pointed with great precision toward the earth's center without knowing the location of the target on the earth's surface. Since the solid angle occupied by the beam will generally be much smaller than the solid angle subtended by the earth, a search will be necessary throughout the latter.

In order to develop an acquisition logic, it is first necessary to define the following parameters governing the system:

ω_t = solid angle beamwidth of the transmitter;

Ω_t = solid angle field of view of the transmitter, i.e., the solid angle representing the uncertainty to which the position of the target is known;

ω_r = instantaneous field of view of the receiver;

Ω_r = total field of view of the receiver, analogous to Ω_t but not necessarily of the same size;

T_a = acquisition time.

It is useful to examine the factors governing the size of each of these.

The transmitter beamwidth ω_t must be selected on the basis of the power available from the laser, the distance to the target, and the required power density at the target. The smaller the power available from the laser, the smaller the beamwidth necessary to realize the same power density at the receiver. Figure 1 is a plot of the maximum range obtainable as a function of the solid angle of the transmitted beam for the following parameters:

Transmitted Power	1 watt
Receiving Aperture	1 ft^2
Receiver Bandwidth	1 Mc
S/N Ratio	5 db
Losses	5 db
Detector Sensitivity	3×10^{-12} dbw (Sensitivity of the best uncooled photomultiplier for 6328Å wavelength.)

This plot is, of course, representative and may be shifted up or down for other values of the above-listed parameters.

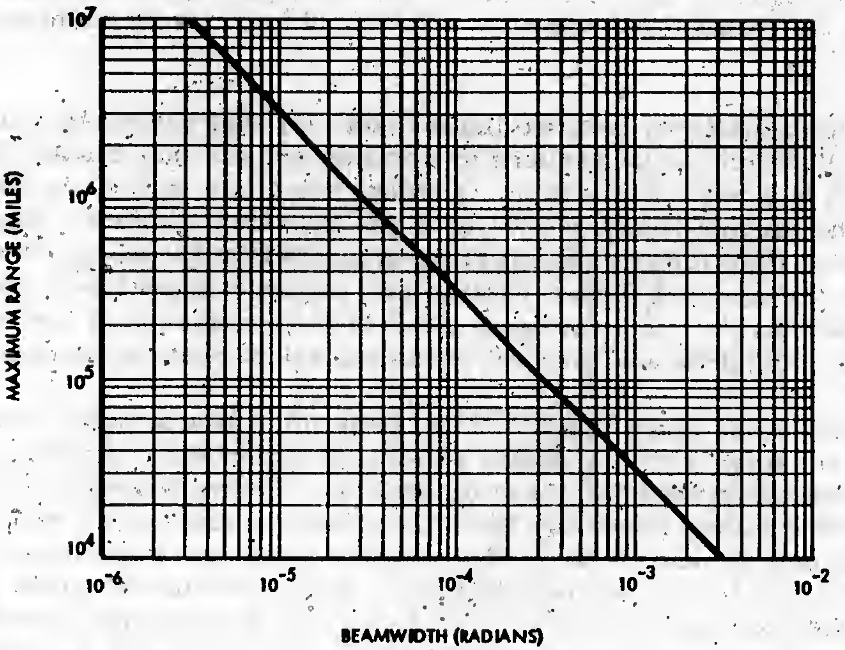

Figure 1. Maximum Range as a Function of Beamwidth

At this stage of development, it is difficult to predict the magnitude of CW power available from lasers in the future. It appears as though power outputs of the order of watts or tens of watts may be obtained but probably not hundreds or thousands of watts. New developments may greatly alter this prediction. Higher output power remains a prime requisite for very long range communications, such as deep space probes, and the course of development must be aimed toward higher powers because beamwidth narrowing to increase power density cannot be exploited ad infinitum.

In addition to the difficulties of aiming such narrow beams, a ground based station must contend with scintillations caused by air turbulence. These scintillations have plagued astronomers since early time and are of the order of one half to two seconds of arc at night and greater in the day. The lower beamwidth limit for a ground based station will therefore be about one to five seconds of arc.

A receiving aperture of one square foot is taken as the size that can conveniently be carried aboard a space vehicle; weight and size of optical telescopes increase rapidly as the aperture is increased beyond this point. Greater ranges might be achieved if a large aperture is used on the ground and is accompanied by a laser of correspondingly greater power. Since it is unlikely that the output power of a single CW laser will be limited by input power considerations, the higher power can be achieved only by ganging several lasers in an array. If the problems associated with arraying lasers can be solved, then greater distances than those indicated in Figure 1 can be obtained. For example, the range can be doubled by increasing the transmitted power by a factor of four, or by doubling the receiving aperture.

Detector sensitivity has two limitations: (1) that caused by noise in the receiver itself which can be reduced by cooling, and (2) that caused by quantum noise which is a basic limitation. Quantum noise is a part of the signal itself and is due to the random times of arrival of the individual photons. Since the energy of a photon at light frequencies is large compared to the energy of a photon at microwave frequencies, fewer photons are required to achieve a given power level. At low power levels, the averaging effect of large numbers of arriving photons is lost and the randomness of arrival of the individual photons becomes important.

Quantum noise rises linearly with bandwidth while detector noise (for photomultipliers) varies with the square root of the bandwidth. Quantum noise, therefore, becomes dominent for large bandwidths. Figure 2 shows plots of the noise level of both quantum noise and detector noise as a function of wavelength for bandwidths of 1 Mc and 10 Mc. The detector noise curve is plotted from data on commercially available photomultipliers. Each point on the curve represents the best available photomultiplier for that particular wavelength. Since the measured points for the detectors do not vary by more than a factor of two from the curve, it is felt that this curve is representative of uncooled detectors.

The total field of view Ω_t of the transmitter is expressed in steradians. It represents the area of uncertainty in the knowledge of the exact location of the target. This information may be obtained, as discussed previously, from high resolution radars, attitude stabilization systems, star trackers, or combinations of these. For the case in which the position of the earth is known, but not the location of the target on the earth, Ω_t becomes the solid angle subtended by the earth with its vertex at the space vehicle.

The receiver beamwidth ω_r, or instantaneous field of view, must be small enough at some time during the acquisition phase to permit pointing the laser transmitter associated with it with sufficient accuracy to ensure illumination of the distant target. Although ω_r may be initially broad, it must eventually assume the same order of magnitude as the transmitter beamwidth on the same end of the communication link.

The total field of view Ω_r of the receiver is selected on the same basis as that of the transmitter and is numerically equal to that of the transmitter at the same location.

The acquisition time T_a is the time allowed to acquire and identify the target, based on convenience and the relative motion between the earth and the space vehicle.

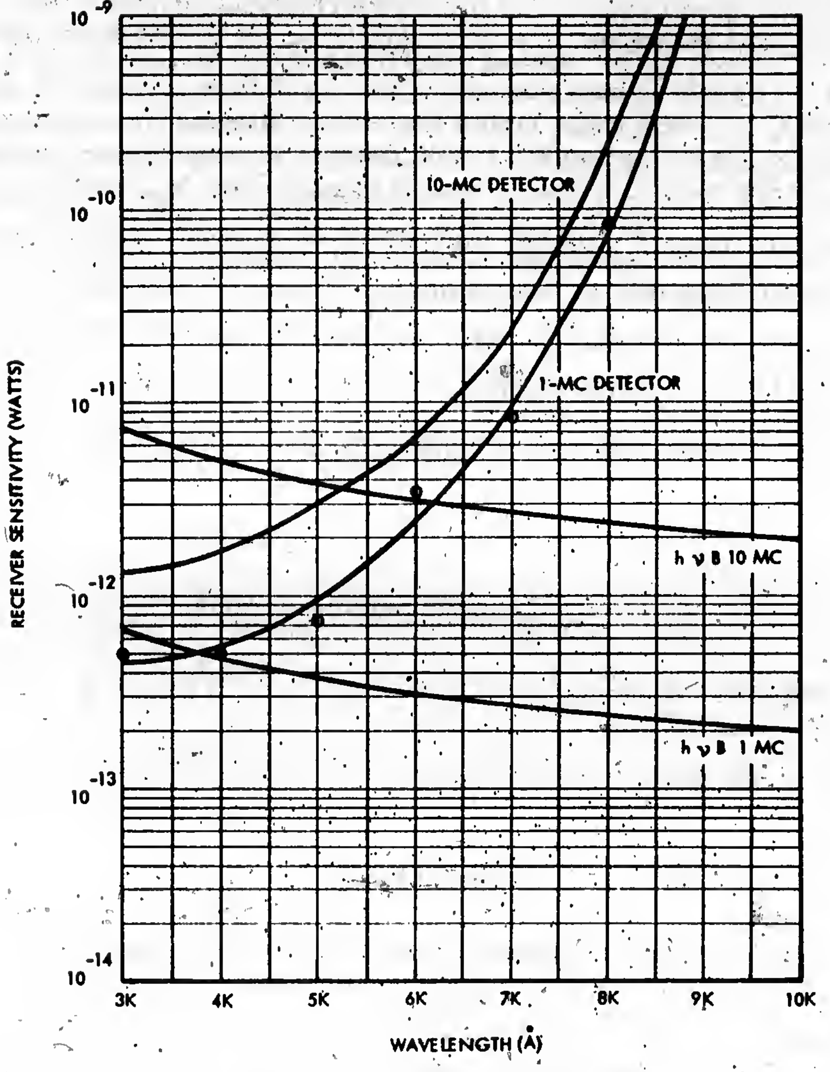

Figure 2. Receiver Sensitivity

Three types of acquisition systems - constant beamwidth, variable field of view, and storage - were studied in detail and are treated at length in the following discussion.

Constant Beamwidth Acquisition System

In this system, the transmitter and the receiver beamwidths remain constant throughout the acquisition cycle. The most general applications of this concept are represented by the relations $\omega_t < \Omega_t$ and $\omega_r < \Omega_r$. In this case the transmitter and the receiver must both scan out their respective field of view in such a manner that, at some time during the period allotted for acquisition, the transmitter and receiver must point at each other. In order to accomplish this, one terminal, say the receiver, scans at a very slow rate R_r, so that it requires the entire acquisition time T_a to scan its field of view Ω_r. At the same time, the other end, the transmitter in this example, scans at a fast rate R_t, so that it scans its entire field of view Ω_t once for each beam position of the receiver. The respective scan rates are then given by

$$R_r = \Omega_r / T_a \text{ steradians/second}$$

and

$$R_t = \Omega_t \Omega_r / T_a \omega_r \text{ steradians/second,}$$

or

$$R_t = R_r (\Omega_t / \omega_r).$$

The dwell time, or the time required for the transmitter beam to cross the receiver, is given by

$$t_d = \omega_t / R_t = T_a \omega_t \omega_r / \Omega_t \Omega_r.$$

A specific acquisition of interest occurs for the case represented by the relations $\omega_t = \Omega_t$ and $\omega_r < \Omega_r$. An example would be the situation in which the position of the space vehicle is known with great accuracy, say within a second of arc. The ground based transmitter could be immediately aimed at the space vehicle, illuminating it at all times. It is then only necessary for the receiver on the vehicle to search out its field of view until the far transmitter is detected. The receiver then begins tracking and directs the beam of its associated transmitter toward the ground to commence communications.

Variable Field Scan System

Optical receiver systems which are not diffraction limited can quite easily be made to have a variable instantaneous field of view. For example, a variable iris, placed at the image plane of the primary lens or mirror, acts as a field stop to change the field of view. This characteristic can be used to advantage in an optical acquisition system to conserve bandwidth and/or acquisition time.

Consider the total field of view of the receiver to be divided into quadrants and the instantaneous field of view to be the solid angle subtended by one of the quadrants. The receiver will examine only four beam positions for the presence of the transmitter and the transmitter will scan its field of view at least once for each receiver beam position. Upon detecting the signal, the receiver narrows its field of view by a factor of two in each direction and again scans. This time its total field of view is the quadrant in which the signal was originally detected. This procedure is repeated until the target position is located within a receiver instantaneous field of view of the desired size. The following relations describe this type of scan and compare it to the fixed field scan.

Let n be the number of beam positions of the fixed field scan in the total field of view; then

$$\omega_r = \Omega_r/n.$$

If, in the variable field scan, Ω_r is successively divided into quadratures as previously described, then

$$\omega_r = \Omega_r/4^m$$

where m is the number of divisions into quadratures. Then

$$n = 4^m$$

and

$$\begin{aligned} \log n &= m \log 4 \\ &= 0.602\ m. \end{aligned}$$

For a number of values of n, m takes the following values:

n	m
10^2	3.32
10^3	4.98
10^4	6.65
10^5	8.30

Since m can take only integer values, the next higher whole number would be used.

On the average, with a fixed field scan system, the target would be located when only half the total n beam positions had been scanned. Even so, it is obvious that the scanning of the field need be accomplished with only a few quadrature divisions compared with the high number of fixed field elements.

The disadvantages of this method are the greater complexity of the scan system due to the zoom nature of the optics and the greater amount of external noise which can enter the receiver with the wide field of view.

Storage Type Acquisition System

The storage type system employs a device such as an image orthicon together with a memory matrix and computer. The entire field of view is imaged on the face of the image orthicon with the position of the target read into the memory. This arrangement is necessary because there may be optical noise sources in the field of view in addition to the target. The image orthicon is an integrating device so that high frequency recognition modulation cannot be placed on the target beam to aid in target discrimination. A low frequency modulation can be placed on the target emission, however, so that a large number of frames will reveal the presence of the identification modulation. The longer acquisition time engendered by this system is offset by the greater sensitivity of the detector due to the long integration time. The disadvantage of this system lies in the need for a memory and computer.

Noise

External noise in an optical system originates from two principal sources, that of the optical signal itself and that from sources which are self-radiating or are reflecting energy from other sources. This noise both deteriorates the sensitivity of the receiver and necessitates discrimination of the target from the other sources.

For an ideal receiver, regardless of frequency, the total noise spectral density is given by the relation

$$\phi(f) = \frac{h\nu}{e^{h\nu/kT} - 1} + h\nu,$$

where $\phi(f)$ = total noise spectral density per cycle of bandwidth,

h = Planck's constant (6.625×10^{-27} erg/sec),

k = Boltzman's constant (1.38×10^{-16} erg/deg),

T = Absolute temperature,

ν = Frequency of radiation.

At optical frequencies, the first term disappears and the $h\nu$ term becomes dominant; at radio or microwave frequencies, $h\nu \ll kT$ and the first term goes to kT.

It is convenient, in the microwave region, to speak in terms of equivalent temperature of the receiver. As a matter of interest, the same calculation can be made at optical frequencies by setting $h\nu = kT$; then

$$T = h\nu/k.$$

For a frequency of 4.75×10^{14} cps (6328Å),

$$T = 22,800° \text{ K}$$

and

$$\text{Noise Figure} = 78.5$$

$$= 19 \text{ db.}$$

This noise figure is that of an ideal receiver and assumes a unity quantum efficiency of the receiver. Even if the receiver is noiseless, its quantum efficiency is poorer than one, and this noise figure will be effectively degraded further.

Other noise entering the receiver is due to bodies radiating or reflecting energy in the spectral region of interest. At night these sources are principally the moon, planets and stars, although the phenomena known as air glow, aurora and zodiacal light add a low-intensity, but detectable background. The earth is a large noise source for reflected light. This factor is important for a space vehicle looking toward the earth.

In daytime, the brightest source by far is the sun, but the atmosphere too is very bright and may be 10^{-7} to 10^{-8} watts at zenith in a 10Å bandwidth and a one-square-foot aperture having a 0.5° field of view.

The spectral distribution of both self-radiating and reflecting extra-terrestrial sources are taken to be that of the sun; this is a fairly good approximation except for reflection from the earth which will be discussed in more detail later.

To assess the problem of noise from extra-terrestrial bodies, it is most convenient to use data compiled by astronomers. Unfortunately for this use, the brightness unit most often employed is that of stellar magnitude. The curves plotted on the graph shown in Figure 3 are used to convert stellar magnitude (abscissa) into power density in watts/meter^2 (ordinate). The upper curve is the observed values of brightness without spectral filtering. The monochromatic nature of laser light makes it possible to reduce greatly the effect of external noise sources by passing all received light through a narrow bandpass filter. The filter greatly attenuates the light from unwanted noise sources by passing only a

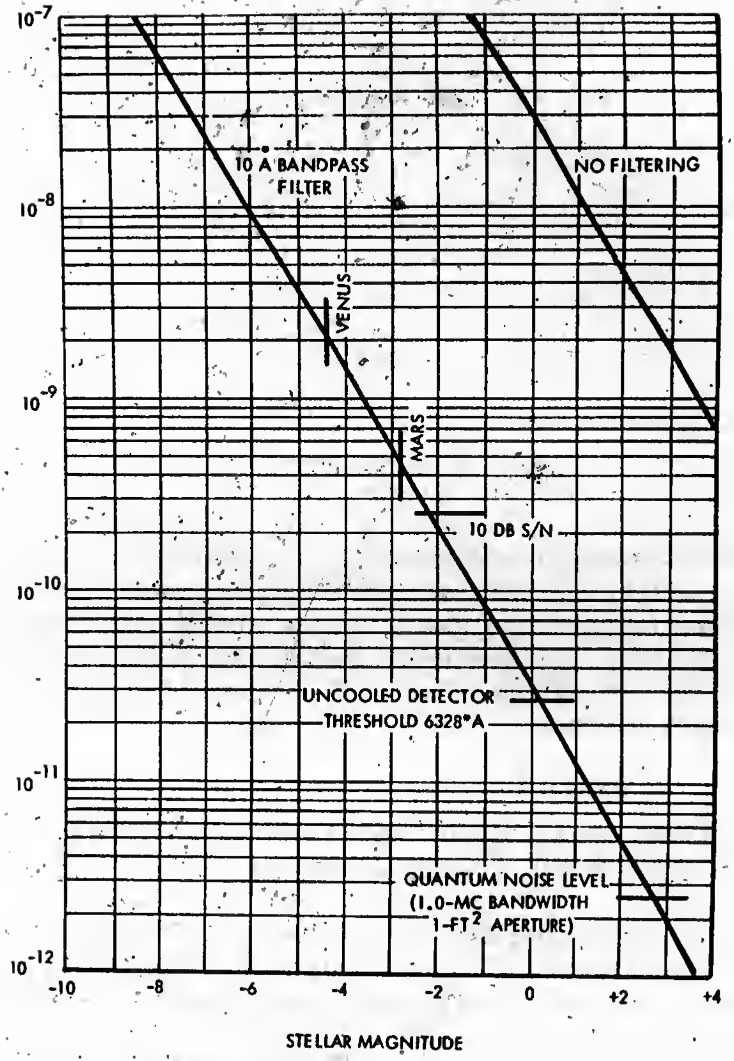

Figure 3. Power Density as a Function of Stellar Magnitude

narrow band of the emitted light with very little attenuation of the laser light. The lower curve in Figure 3 is a plot of the noise passing through a 10Å bandpass filter. A filter of this width can be obtained without too many problems resulting from frequency shifts due to temperature variations. Even narrower filters can be built, but these usually require a controlled temperature environment. It is seen that this filter reduces noise by a factor of 10^3.

Plotted on this curve is the stellar magnitude of the moon, Venus and Mars. Also plotted is the detector threshold for a one-square-foot aperture, which is at a stellar magnitude of 0 order when the 10Å filter is used. The only sources in the sky brighter than this magnitude, hence the only sources which will be seen through an aperture of 1 square foot, are:

Object	Stellar Magnitude
Sun	-26.7
Moon	-12.6
Venus	- 4.4
Mars	- 2.8
Jupiter	- 2.5
Mercury	- 1.2
Saturn	- 0.4
Sirius	- 1.58
Canopus	- 0.86

From these data the brightness of the moon appears to be too great to make optical communications possible; however, this brightness is from the whole moon and can be cut down considerably if only a portion of the moon is viewed. The sun is so bright that communications are not feasible if it is part of the background.

The brightness of the earth is rather difficult to define since there is so little data available on it. Venus owes its brightness to its ever present cloud cover which reflects equally over its surface. The earth, on the other hand, is composed of large masses of water and land. The water reflectivity depends on the surface roughness and will vary from 0.03 to 0.40. Land reflectivity depends greatly on the region from which the reflections occur, the season of the year, and the cloud cover in the field of view. Because the space vehicle will use a receiver with a narrow field of view, it will look at only a small portion of the earth at a time and will receive reflected solar energy which varies greatly with location. The approximate reflectivities of various surfaces listed below are taken in the visible region and are not valid in the near infrared since reflection from vegetation increases greatly in that spectral region.

Substance	Albedo
Fresh Snow	0.80
Old Snow	0.40 (approx.)
Grass	0.1 - 0.23
Rock	0.12 - 0.15
Dry Earth	0.14
Wet Earth	0.08 - 0.09
Water (Sea)	0.03 - 0.40
Forests	0.05
Deserts	0.25
Clouds	0.5 - 0.75

The large variations of albedo might make it worthwhile to locate ground based stations in regions of low albedo. For example, if operation is in the visible region, the most suitable area might be forest areas, but these would be among the worst locations for near infrared operation.

The average albedo for the earth as a whole, if it occupies the entire field of view, ranges from 0.32 to 0.52 which varies with the season and greatly with the amount of cloud cover.

The effect of these noise sources on the acquisition and tracking function of an optical communications system is twofold. First, it raises the noise level of the receivers, thus necessitating a greater signal level at the receiver. Secondly, the sources require a means of target discrimination so that a false target is not acquired and tracked.

Spectral filtering, as previously mentioned, is helpful in greatly reducing the external noise problem, but still leaves a few sources at least as bright as the laser signal. In order to provide discrimination from these sources, it will be necessary to place identification modulation on the signal during the acquisition phase. The received signal can then be passed through a narrow bandpass filter so that it can be recognized.

The Tracking Problem

The problem of tracking with very narrow beamwidths centers about the need for precision. In order to operate a successful optical communications system, it is necessary to track within a small part of the beamwidth of the transmitter. Precision of tracking, therefore, must be of the order of 0.1 to 1.0 second of arc. In addition, the use of extremely narrow beams tends to accentuate the refraction and scintillation effects of the atmosphere. Compensation for these effects may require tracking which is rapid as well as precise.

Tracking Rates between Earth Station and Space Vehicle

The tracking rates associated with relative tangential motion between the opposite ends of the communication system may vary widely depending upon the precise geometry of the system. For example, the tracking rates may vary from 0.06 radian/sec for an earth satellite as viewed from the ground to 7×10^{-5} radian/sec, the rate given by the angular velocity of the earth's rotation. Higher rates than these might be encountered between two earth satellites or space vehicles.

Error Signal Generation

There are a number of error signal generation methods used today which are capable of the high degree of accuracy necessary for a laser communication system. The various types of devices used for applications, such as precision star tracking, are well covered in the literature and will not be further discussed here. One characteristic none of these has, however, is a clear-cut capability for dual use in both acquisition and tracking. A high-speed beam deflector which works upon the application of an electrical signal to steer or deflect the beam is well suited to such a dual role. This beam deflector, if it has the capability of both relatively large and small, precise deflections can be used both for acquisition scanning and for error signal generation in a conical scan mode. This type of beam deflector, if it can be made inertialess as discussed later, appears to be the best suited to perform this function.

Bradley and Transit Time Errors

If the laser transmitter and distant receiver are moving with respect to each other, the transmitter will appear to be at an angle α from its true position given by

$$\alpha = v_p/c,$$

where v_p is the relative velocity component perpendicular to the line of sight between transmitter and receiver, and c is the velocity of light. This deviation is called the "Bradley error" or "transit time error" of the communications link.

It is convenient to remember that a relative velocity of 1 mile/sec results in a Bradley error of 1.107 second of arc. Thus, relative tangential velocity in miles per second is approximately equal to the Bradley error in seconds of arc.

The orders of magnitude of Bradley error that would be encountered in optical communications links are:

Communication Link	Bradley Error (seconds of arc)
Earth to Earth Satellite	1 to 7
Earth Satellite to Lunar Satellite	3 to 8
Earth to Planetary Probe	5 to 25

It is seen that these errors are significant when considering beamwidths less than 10 seconds of arc.

Atmospheric Effects

When one portion of an optical communications system is located within the atmosphere, its propagation path is subject to losses and perturbations caused principally by scattering of the light by particles and absorption by the various gases. The subject of scattering has been under study, both theoretical and experimental, for quite some time. Rayleigh, in 1871, made the first quantitative study of scattering by small particles. Scattering by particles much smaller than a wavelength is commonly called "Rayleigh scattering" and shows a wavelength dependence such that the scattered intensity is proportional to λ^{-4}. Thus, when white light is scattered by smoke or haze, it has a bluish color as does the sky. The scattering caused by particles which are about the same size as a wavelength is known as "Mie scattering" and is the most complicated of the scattering phenomena. Experimental evidence on this type of scattering indicates a dependence of scattered intensity on wavelength of $\lambda^{-1.3}$ to $\lambda^{-0.7}$. The discrepancies are undoubtedly due to variations in the scattering atmospheres which were studied. When the particles are much larger than a wavelength, no dependence of scattered intensity on wavelength is observed so that the scattered light is white as observed on clouds illuminated by the sun.

A great deal of work has been done on atmospheric absorption in the visible and infrared portions of the spectrum but studies with high resolution were made only recently. The constituents of the atmosphere which cause absorption in the visible and near infrared include oxygen, water vapor, and carbon dioxide. The absorption is generally due to molecular resonance lines many of which are observable only under high resolution because they are quite narrow.

Before an operating wavelength is selected, it is necessary to examine atmospheric absorption in the region immediately surrounding the wavelength of interest for absorption lines. The He-Ne line at 6328Å is in a favorable region with no significant absorption lines in its vicinity; thus, it will suffer no measurable absorption. However, the wavelength 1.152μ lies in the vicinity of five water vapor lines, all of which contribute to absorption at atmospheric pressure. Because reduced pressure narrows the lines, only two of those absorption lines contribute to absorption at high altitudes. The ruby wavelength is in a region of a number of lines of both oxygen and water vapor so that heating it, thus causing a change in wavelength, subjects it to a variety of absorption conditions. Further into the infrared region, where many laser wavelengths lie, the number of absorption lines increases and, in addition, molecular band structure becomes important so that only a few good "windows" exist. The 3.5 and 3.99μ lines of helium-xenon are infrared lines which exhibit very low atmospheric attenuations.

Losses, or perturbations, caused by refractive differences in an active atmosphere constitute the other troublesome factor affecting optical communications. The atmosphere is characterized by localized variations in its index of refraction caused by temperature gradients. These variations, or refractive "cells", are caused by turbulence in the air and are frequently associated with regions of high wind velocity and/or wind shear such as that found in the vicinity of a jet stream.

At any instant of time, the effect of these refractive cells is to produce bending and some break-up of a beam traversing them. This beam "dancing", or scintillation, causes amplitude variations at the receiver.

Several investigations were recently made of refraction effects over horizontal paths. Unfortunately, these studies are not very useful for a vertical propagation path because the air motions are so different. These studies will have some limited application for propagation angles near the horizon.

Vertical propagation studies have been made by observing stars or lights on aircraft as sources; however, no known studies have been made to date employing coherent light over vertical paths. The studies which have been made indicate that the major scintillation effects originate in a layer at an altitude of 40,000 to 50,000 feet. Good correlation was obtained between the altitude of the scintillating layer and wind speeds aloft. The amount of beam deflection, or scintillation, for vertical paths amounts to 0.5 second of arc for good seeing conditions to about 2 seconds of arc for poor conditions. The spectrum of the noise induced by scintillation appears to have an upper limit of 500 to 1000 cps with a general decrease in amplitude from 1 to 500 cps. The frequency generally increases as the angle from zenith increases. There is some evidence that signal fluctuations caused by the atmosphere scintillations decrease as the observing aperture increases, thus indicating an averaging effect over a distance of a foot or two.

In contrast, the recent work done on horizontal paths indicates beam deviations as high as 5 seconds of arc with a frequency spectrum generally decreasing from 1 to 100 cps but peaking at approximately 20 cps.

One other atmospheric effect which should be mentioned for the sake of completeness is the various phenomena which can be grouped under the term atmospheric emission; this includes air glow, aurora, and thermal emission. These effects, however, are quite weak in intensity and are of no importance when a narrow band spectral filter is used on the receiver.

System Configurations

The following discussion concerns the equipment needed to perform the precision tracking and acquisition tasks treated in the preceding paragraphs. For convenience, the discussion is divided according to the following basic equipment components: beam deflector, modulator, optical system, and detector.

Beam Deflectors

Both the acquisition and tracking modes of operation depend on beam steering or deflection. Certain acquisition schemes need deflection rates up to 1 Mc, while requirements for both acquisition and tracking call for a high degree of precision. Certainly, the beam deflector must be capable of pointing the beam to a required angle to within one half of the beamwidth or less. High precision deflection can be obtained by purely mechanical means such as a moving mirror but high deflection rates require accelerations too high for this arrangement.

Another possible arrangement, which could be more satisfactory if successfully implemented, would be a beam deflection system which is electronically controlled by the application of electrical energy. In order to obtain beam deflection which is dependent upon electrical power, and not on mechanical displacement, it is necessary to find a means to alter the phase front of the light beam. One method might be to introduce a variable index of refraction gradient in a plane perpendicular to the direction of propagation of energy. The electro-optic effect in crystals, the elasto-optic effect in crystals or plastics, the Kerr effect in certain liquids, or the Cotton-Mouton magneto-optic effect are all possible means to accomplish beam deflection by altering the index of refraction.

Beam deflection has been accomplished using both the electro-optic and elasto-optic effects. Of the two, the electro-optic effect in potassium dihydrogen phosphate (KDP) has shown the greatest promise. Beam deflection through an angle of one half degree or so has been demonstrated with rates in the kilocycle region. Neither the maximum possible angles nor rates are represented by these figures. The state of the art in these devices is still so new that many questions are still unanswered. For example, little is yet known about deflection precision, temperature effects, or effects of crystal anisotropies. It will be some time before a sufficiently complete evaluation can be made of this class of device to determine suitability for the acquisition and tracking tasks in an optical communications system.

Optical System

The optical system for optical communications must serve the dual function of collimating the transmitter beam to the desired size and to efficiently collect energy for the receiver from the desired field of view. It is possible, therefore, to have two separate optical systems, one to transmit and one to receive, or to combine the two functions into a single optical system.

The coherent nature of laser light makes it possible to employ optics for transmitting which approach the diffraction limited value in size. The necessary diameter of optics under these conditions is given by the following relation

$$d = 1.22\lambda/\theta,$$

where
d = aperture diameter in centimeters,
λ = wavelength in centimeters,
θ = beamwidth in radians.

A one-second beamwidth, then, requires an aperture diameter of 17 cm. The sizes of the receiving optics, on the other hand, are not selected on the basis of diffraction limiting but rather on the maximum light gathering power for allowable size and weight. Optical telescopes of large aperture are usually of the reflecting, rather than refracting, type since weight and distortion are less. Even reflecting telescopes become large as the aperture is increased because of the longer focal lengths necessary and will have a practical limitation for use in space vehicles.

The extreme accuracy requirements of the optical system calls for a very rigid system. For example, the maximum allowable deflection of the focal point will be of the order of 4×10^{-4} inch or less. A way to meet this requirement as well as to decrease size, and perhaps weight, is to shorten the optical system by folding. The folded optical configuration which has proven most satisfactory through the years is the Cassegrain system.

An optical receiver employing Cassegrain optics is shown in Figure 4. The primary mirror is the collecting aperture and has a parabolic surface. The secondary mirror is placed in front of the primary mirror at such a distance that aperture blockage caused by it is acceptable. The secondary mirror has a reflecting surface facing the primary mirror which is a hyperboloid of revolution. One focal point of the hyperbola is coincident with the focal point of the primary mirror while the other focal point lies within the system as shown. Following the focal point is a lens of short focal length, such as an eyepiece lens for a telescope, which has its focal point coincident with that of the hyperboloid. The function of this lens is to re-collimate the received signal before it is passed into the beam deflector. Between the lens and the beam deflector is a convenient point to place a narrow band interference filter to reject all noise but that near the desired wavelength. The beam deflector is then followed by a detector which will be a photomultiplier if the radiation is in the visible or near infrared portion of the spectrum.

Operation of this receiver is as follows. Incoming light energy from a wide field of view is collected by the primary mirror. This mirror, in conjunction with the secondary mirror, focusses that energy on the focal plane. The focal plane lies perpendicular to the axis of the optics and is located at the position of the indicated focal point. The beam deflector then acts as a variable position field stop and selects energy from only a tiny area to be transmitted to the detector. The beam deflector, therefore, imparts to the system the capability of looking at only a small field of view at a time and of changing that field at will.

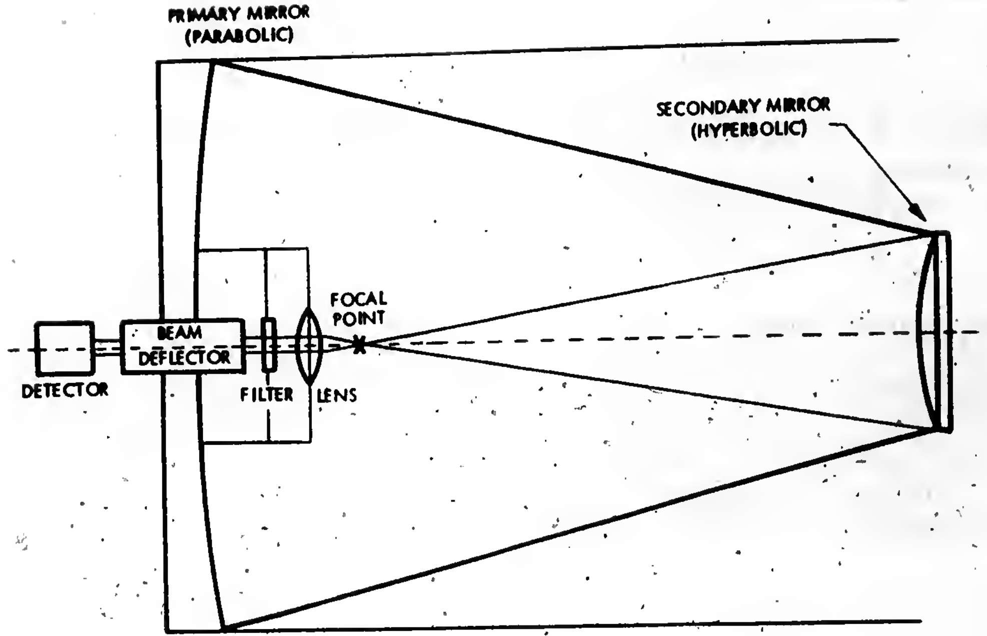

Figure 4. Cassegrainian Optical System

Should it be desirable to change the size of the field of view, as in the variable field system, the focal point of the lens need only be moved so that it is non-coincident with that of the secondary mirror.

Size and weight reduction, as well as the need for two identical beam deflectors if separate optics are used for transmitting and receiving, makes the possibility of combining the two functions in a single optical system attractive. The difficulty encountered with this combined system is that half the transmitted and received signal must be lost in the beam splitter and isolation between transmitter and receiver will be almost impossible to obtain. The latter problem cannot be solved by time sharing because tracking requirements will necessitate a constantly transmitted signal.

There is one scheme, however, which may avoid both of these problems; that is, to make use of the fact that lasers are available which operate at a number of different wavelengths. If the transmitter and receiver were to operate with a common optics system but at different frequencies, good isolation could be obtained through spectral filtering in the receiver. If this were done, the other difficulty, that of power loss in the beam splitter, could also be overcome by using a dichroic beam splitter. A dichroic mirror or beam splitter is one that is made to transmit at one wavelength and reflect at the other. Thus, it is possible, through a two color scheme, to eliminate both major difficulties encountered in using common optics for transmitting and receiving. A new problem which may arise, however, is one of dispersion. If anything in the path of the beam, such as atmospheric refraction cells, has a different index of refraction for the two wavelengths used, the possibility exists that they will follow different paths. If the path directions for the two wavelengths should deviate by more than one half the beamwidth, the system could be rendered inoperable.

Detectors

The choice of the optical detector will depend upon the wavelengths, convenience, and bandwidth requirements of the particular system under study. In the visible and near infrared, the detector choice will be a photomultiplier while operation further into the infrared will necessitate employment of one of the various available solid-state detectors.

Superheterodyne detection will probably not be employed for this type of communication system because of Doppler shift. For example, if the relative velocity between the two ends of the communication path is 1000 miles/hour (a somewhat conservative value) the output frequency of the superheterodyne detector due to Doppler shift is 700 Mc. Therefore, although a superheterodyne optical system could be valuable for extracting Doppler information, it would be a hindrance for a communications system whose needs are better served by a video detector.

Conclusions

Long-range communication systems based on the laser as the source of radiated power will probably prove feasible for broadband operation. It is unlikely, however, that optical systems will successfully compete with systems operating at microwave or lower frequencies when narrow bandwidths are used.

The problems of acquisition and tracking can be solved assuming the availability of good coarse position information, although better components must be developed. The needed components are precision high-speed beam deflectors, wide-band modulators and light, compact optical systems capable of withstanding launch and space environments without losing alignment. High-power CW lasers are also needed, together with the means of paralleling several lasers for increased radiated power. All of these components are currently the subject of intensive study in laboratories throughout the country and all appear to be technically feasible.

"TOLO"

TECHNICAL ORDER LIBRARY IN ORBIT
(TECHNICAL ANALYSIS)

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Martin Company
Denver, Colorado

SUMMARY:

This paper is a follow-on technical analysis of "The Technical Order Library in Orbit",[1] a paper written by Mr. Gene W. Winz, Titan I Technical Writing Chief, and presented to our Denver Division and corporate management last summer. In his paper, Mr. Winz proposed three alternate methods of providing technical order support to a manned orbital space station. These were: ground station-to-space station, orbiting data station, and on-board technical orders (in microform). His conclusion recommended the ground station-to-space station approach.

Mr. Moravec and I have elected to expand on all three of the philosophies and present possible methods of accomplishing; data storage, transmission, receipt, and display; not only for manned orbital space stations, but also for manned space vehicles.

NEED FOR A LIBRARY:

With the advent of manned orbiting space stations and space vehicles, more and more emphasis is placed on the reliability of the vehicle. Generally, space experts agree that the chance for survivability of a space mission could be maximized by in-flight maintenance. Back-up systems and redundancy can help; however, there would be a space and weight limitation to redundancy. A significant increase in the morale of the crew could be obtained if they have the self-contained ability to maintain, inspect and repair malfunctioning equipment.

Due to the quantity and complexity of the equipment required on board the station or vehicle, information for proper operation would be required in addition to that of maintenance and repair. Since it would not be practical to store spares at the chassis or black box level, our Technical Publications Department has estimated that the necessary technical manuals required for maintenance at the piece part level of the type of systems that might be utilized in a space vehicle would require a library that would weigh approximately 300 pounds in bound manual form. The weight and volume of this information could be reduced by utilization of methods such as micro-filming and video-audio transmission and tape recording.

REQUIREMENTS OF A LIBRARY:

Since the library would be used primarily in case of trouble, information must be available with a minimum of time involved. This would require an efficient indexing system in conjunction with a speedy method for sorting out the required data. The library would have to be capable of being updated or changed rapidly and have extremely high reliability. Means of displaying the information to several areas within the space vehicle simultaneously would also be a desireable capability.

LOCATION OF THE LIBRARY:

There are several approaches to the problem of maintaining a library for a space station or vehicle. These are as follows:

1. It could be maintained in a ground station or stations.
2. It could be maintained in an orbiting satellite.
3. It could be maintained on board the vehicle.

1. GROUND STATION LIBRARY:

A complete library could easily be maintained on the ground and stored for rapid search and retrieval in a Mechanized Data[2] system such as FASTI or A-VIS.

Fast Access to Systems Technical Information, FASTI,[3] was developed by General Dynamics/Astronautics for the Atlas Weapon System. FASTI is a program covering the preparation, retrieval and display of technical data. With FASTI, maintenance personnel can isolate a malfunction in 15 minutes, compared with five hours required with ordinary troubleshooting methods - a time saving of 95%.

The preparation phase corresponds to the creation of a troubleshooting chart by a technical writer. Time and function oriented logic diagrams are developed for each circuit. These are then translated into Boolean algebra equations for use in an IBM 7094 computer. The computer does the decision making by comparing normal indications with those for malfunctions by actually failing a component in Boolean algebra.

The output of the computer includes a sequential print-out of the most logical causes for each malfunction.

Standard Recordak microfilm retrieval and display equipment is used. When a system indicator reveals a malfunction, the corresponding microfilm index number is determined by the operator. This index number is inserted into the Recordak machine by the operator through a pushbutton command keyboard. The microfilm record related to the malfunction is displayed on the screen. If desired, the operator may obtain a print of the display. If the malfunction is not identified in the first display, it calls forth the index number of another display which corresponds to the next most logical level for the malfunction. This procedure is continued until the malfunction is located.

A-VIS is an experiment in the area of mechanized retrieval being conducted by the Army Materiel Command. This experiment is a follow-on feasibility study (MODAPS)[4] resulting from an earlier investigation conducted by Bell Telephone Laboratories and Western Electric Company. The study indicates the definite feasibility of developing an automated information system featuring audio-visual concepts in support of requirements for operation and maintenance data.

A-VIS consists of a shoulder height console with an automated slide projector, a synchronized magnetic tape, and a pair of earphones. The aural instructions provide the user with step-by-step instructions while the projector displays an illustration of the operation. Changes to data are made by substituting a new tape and replacing slides in the film chip magazine.

A-VIS is designed to provide rapid retrieval and display of maintenance data including schematics, checklists, troubleshooting instructions, adjustments and other related data. It also provides automation of certain maintenance functions such as reporting unsatisfactory equipment, recording data obtained during periodic checks, and requisitioning spare parts.

The information system consists of two rear projection viewers, a mechanized film-chip magazine with a capacity of 10,240 microfilm chips, and automated data collection units. The operations console has two film-chip magazines; an "A" screen magazine for regular page size projections, and a "B" screen magazine for large foldout type illustrations, each having individual push-button controls. The film cartridge transport unit is mechanically positioned in the projector for vertical and horizontal movement. The console also has a hard copy capability so it does not tie up the machine.

When the need for information would be required by the space vehicle, the vehicle would transmit a specific code number or numbers to the library corresponding to an indexing system for the technical areas covered. The signal received at the ground library would be decoded and sent through the memory unit of the computer which in turn would sort out the requested data and transmit it back to the requesting vehicle or station in the form of a video-audio signal.

Basically no audio conversation would be required with the ground crew, unless additional information not contained within the library would be needed. If necessary, a pre-recorded audio tape within the library similar to that in "A-VIS" could be transmitted along with the video signal. In addition, should it be necessary, a live video-audio conference could be arranged between the vehicle crew and engineering specialists on earth.

The information received in the vehicle could be viewed directly on a master and slave monitors of a closed circuit TV system and simultaneously tape recorded for future reference, allowing it to be played back through the TV system as required. Thus, personnel would receive the most up-to-date instructions for correcting the problem with a minimum of preoccupation. A recorder with a "stop tape" capability would be utilized to greatly increase the storage capacity of printed data. The "stop tape" feature would allow one frame to be viewed at a time, making it possible to record a different page of text or illustration on each frame. The tape transport would be stopped and the individual frame scanned by a revolving head, transmitting the image to a master and/or slave monitor screens for viewing.

By using slow scan TV, with a scanning rate of 1/2 frame per second, and retaining the same resolution as commercial TV, a conventional video tape recorder could be slowed down to have a recording time 60 times longer than present off-the-shelf units.

An audio warning system, such as the one used on the B-58 Hustler should also be considered in the vehicle. As problems occur, a pre-recorded audio tape describing the corrective steps to repair the malfunction would be automatically played over the PA system. If more than one failure occurs, the corrective action for the incidence of primary importance would be played first. Once corrective action has been taken, the tape playback would advance to the next major failure.

The vehicle closed circuit TV system would be quite an asset in conjunction with the audio warning system. By placing TV monitors in hazardous or remote areas such as the exterior of the vehicle, or fuel areas, monitoring could be accomplished without exposing the crew to unnecessary dangers.

Depending upon the mission of the space vehicle, the transmission requirements for communications between the vehicle and the ground station could be quite complex. Since the position would be constantly changing between the ground station and the space station or vehicle, active communication relay satellites and/or additional ground stations would be required to maintain constant communications. The air-to-ground communications would be the weakest link in the library system and thus must be extremely reliable. This could be accomplished with redundant and/or easily repairable equipment.

Due to the possibility of failures in the vehicle power supply or in the communications system itself, a small library would have to be maintained on the vehicle. This library should be independent of the communication and display systems and the vehicle power supply which are commonly used as part of the master library system. This "emergency" library would contain data concerning the power supply, communications, and abort systems stored on 16mm film. The viewer could be hand held or case mounted[5] and contain its own light source and battery supply. A redundant power source in the airborne transmitter and receiver should also be considered.

The airborne and ground receiver sensitivity and transmitter output will be dependent on bandwidth, airborne power source available, antenna gain, and distance. Based on the limitations of the vehicle, airborne communications would be restricted to the amount of transmitted power output and the antenna configuration as compared to the ground station. With the great distances involved in space travel and because of the "hole in the sky" for radio transmission (1 to 10 gigacycles)* ground stations such as Telstar's Andover, Maine station, elaborate antenna and tracking systems would be required for reception of the signals transmitted from the space vehicle. Because of the lower antenna gain on the vehicle, the ground station must transmit a high power output. The Andover station transmits RF at 2 kilowatts as compared to the received signal of approximately 1 picowatt.+ The satellite transmitter output would be approximately 2 watts with the received signal at about 1 nanowatt.**

For transmission beyond the earth's atmosphere, the carrier frequencies are somewhat limited to a band between 1 and 10 gigacycles. Below 1 gigacycle (1 KMC), cosmic noise increases in amplitude thus reducing the usable signal-to-noise ratio. Above 10 gigacycles (10 KMC), galactic noise and water vapor absorption increase RF attenuation and thus reduce the usable signal-to-noise ratio.

The communication system bandwidth, operating within the range of the abovementioned frequencies, would be dependent upon the type of modulation that would be used. The airborne receiver need only detect the video and audio transmission from the library, whether it be from the ground or another satellite. By using a slow scan TV system which transmits and receives one frame every 2 seconds, the bandwidth would be 150 kilocycles per second.

* 1 gigacycle = 10^9 cycles per second

+ 1 picowatt = 10^{-12} watts

**1 nanowatt = 10^{-9} watts

The audio bandwidth in turn, would be about 4 kilocycles per second. Utilizing AM single side band modulation, the airborne receiver bandwidth could be as narrow as 160 kilocycles per second. The type of coding, such as binary or selective tone frequencies that could be used with the library indexing systems airborne transmitter also affect the bandwidth.

Each space mission will require its own study to determine the characteristics peculiar to it and the necessary design paramiters required to do the job.

Due to the earth's rotation, several ground stations would be required to maintain constant communication with the space vehicle. However, it would not be necessary to have a complete library at every ground station. These stations could communicate with a central library by means of landlines, microwave repeaters, or relay via satellite repeaters. Communications between the vehicle and earth could also be relayed via orbiting communications satellites instead of multiple ground stations. (See Figure 1)

If no library were involved with the space vehicle, the communication system might consist only of an audio link. However, since television is used in the previous example, the addition of a camera in the vehicle could be used for monitoring critical functions and transmitting this information back to earth for further and immediate evaluation and assistance from engineering and scientific specialist teams. During extended trips, the TV communications systems could also be used for leisure time entertainment such as televised mail or pictures of the crew members and their families along with audio exchange of information.

2. ORBITING SATELLITE LIBRARY:

A second method of supporting a library for a space vehicle would be to place it in an orbit around the earth or some other planet. The space vehicle would be similarly equipped as in the previous example to communicate with the satellite. However, in this instance the satellite's library storage, retrieval and transmission equipment would be completely automated. By appropriate orbital orientation of the library with respect to the space vehicle, the distance for transmission between the two could be controlled to a degree, easing the problem of the propagation requirements. In order for the vehicle to keep in constant communications with the satellite and the satellite maintain constant communications with earth, several satellites would be required. We must realize that the increase of components would result in decreased system reliability.

The ground station must communicate with the satellite for updating of the library in addition to relaying of any communications with the vehicle. As mentioned earlier, the communications equipment on the vehicle and satellite are not as powerful or sensitive as those on the ground. Because of the distances involved for space missions, active relay satellites could be required at various orbiting distances. Since the satellite would be unmanned and the equipment automated, the repair capability would be limited and fully dependent upon redundant components. (See Figure 2)

3. ON BOARD LIBRARY:

As mentioned earlier, a complete library with all the necessary manuals in bulk form on board the vehicle would be impractical because of the volume required. Also, thumbing through them looking for the required information

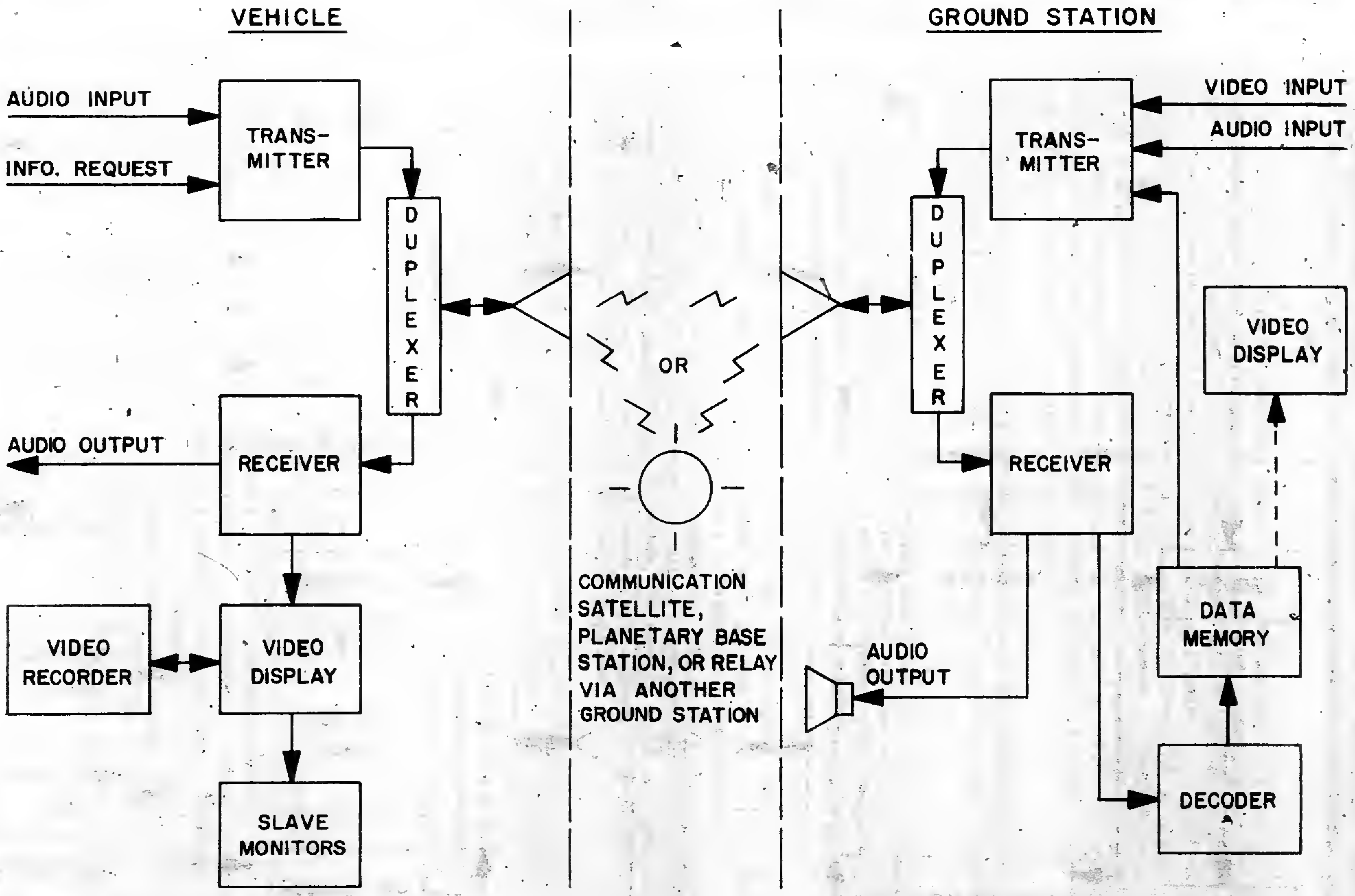

FIGURE I-BLOCK DIAGRAM FOR A SYSTEM WHICH CONTAINS A LIBRARY IN A GROUND STATION

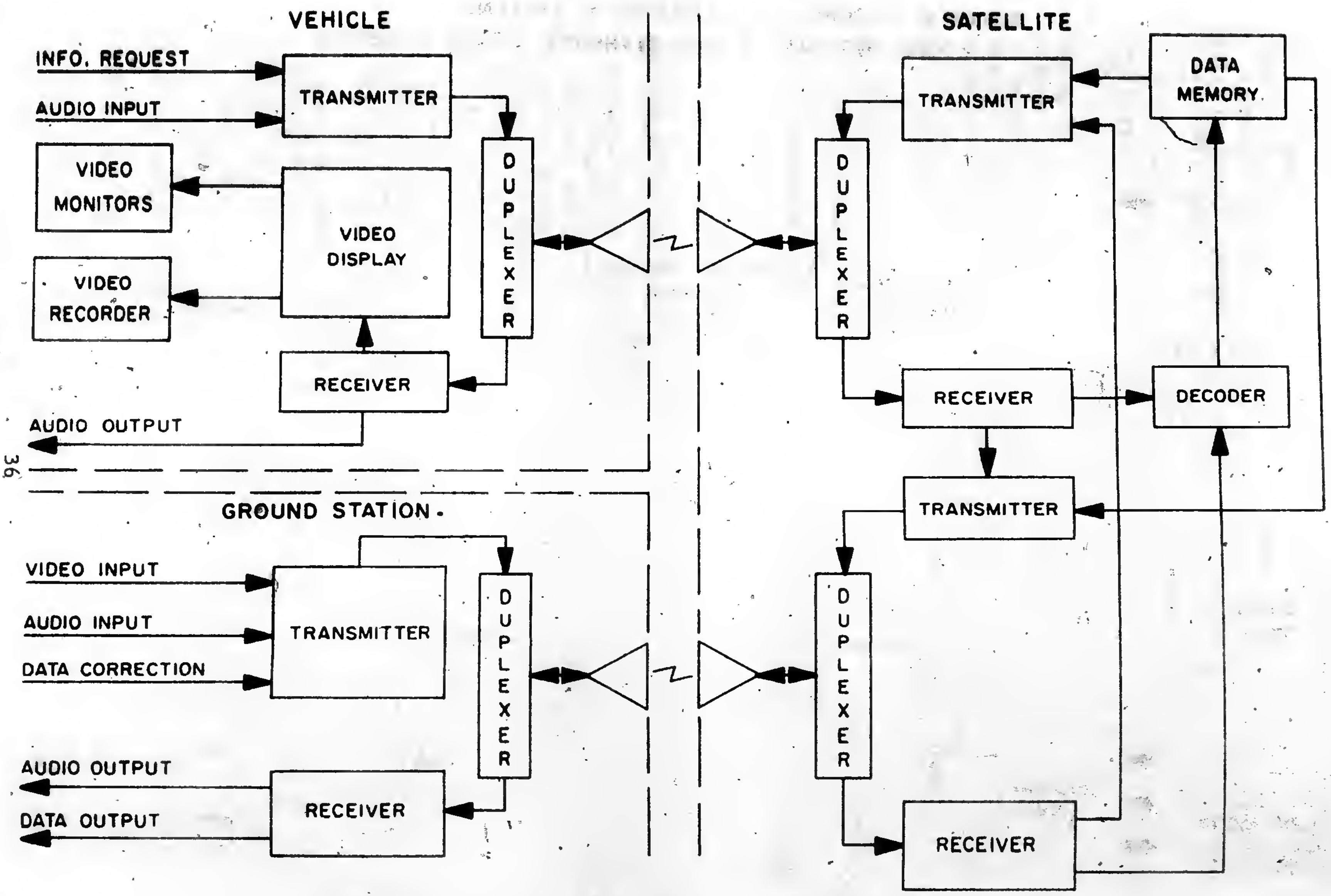

FIGURE 2 – BLOCK DIAGRAM FOR A SYSTEM WHICH CONTAINS A LIBRARY IN A SATELLITE

would be quite time consuming. These problems would be alleviated considerably by the utilization of microfilm or photochromic micro images[6] (Micro Dot) for storage of the T.O. library.

The National Cash Register Company, Hawthorne, California, has developed a device employing a new technique for the storage and dissemination of micro-documents. The technique, known as photochromic micro images (Micro-Dot) makes it possible to make tremendous reductions ultimately recorded on high definition film. Linear reductions of 200:1 make it possible to record a 300-page book within a square inch of film. Expressed in another way, the end item is a photochromic plate 3 x 5 inches capable of recording 2,625 micro-images. With photochromic micro images an entire set of Titan II technical manuals, involving approximately 21,000 pages, could be recorded on 10 3 x 5 cards which could easily fit in a shirt pocket.

In the spring of 1963, NCR successfully demonstrated the feasibility of Micro-Dot under a contract with the Navy Bureau of Supplies and Accounts, Long Beach, California.

A drawback to this would be that information could not be conveniently updated. Corrected information would have to be transmitted to the vehicle and new films made and corrections spliced into existing film. This would require other items such as cameras, chemicals, projectors, editor/splicers, etc., in addition to the vehicle communications equipment.

Let us assume that a problem occurs, corrective information would be obtained from the "On Board" library, and displayed on a projector or similar device. Several groups of people may require this information simultaneously at different locations in the vehicle. Reproducing and delivering this data could be accomplished with the closed circuit TV system mentioned earlier or by using machines such as Verifax, Recordak, Polaroid, etc., to make hard print-out copies. However, storage of photographic papers, chemicals, and film on board the vehicle would pose an additional problem.

The major advantage to this system would be that all of the basic library information would be self-contained within the vehicle and not basically dependent on an external source. (See Figure 3)

In the event of a failure in the power system on the vehicle, the video recording and playback equipment and audio warning systems should have their own secondary power source.

CONCLUSION:

One of the main considerations for determining the type of library system would be the qualifications of the people on board the vehicle. In the beginning, manned space shot crews will consist of two or three astronauts whose primary function will be piloting of the vehicle and various life science experiments. The repair of malfunctioning equipment could be considered as secondary. These people probably would not be trained electronic or mechanical engineers, chemists, or technicians who would be familiar with all the equipment on board. Therefore, they might have to rely upon others to expedite repairs. Consequently Approach Number 1, having the library on the ground, would be the most workable. A small library would still have to be maintained on board; however, technical assistance from the designers and scientists on the ground would be feasable and advantageous.

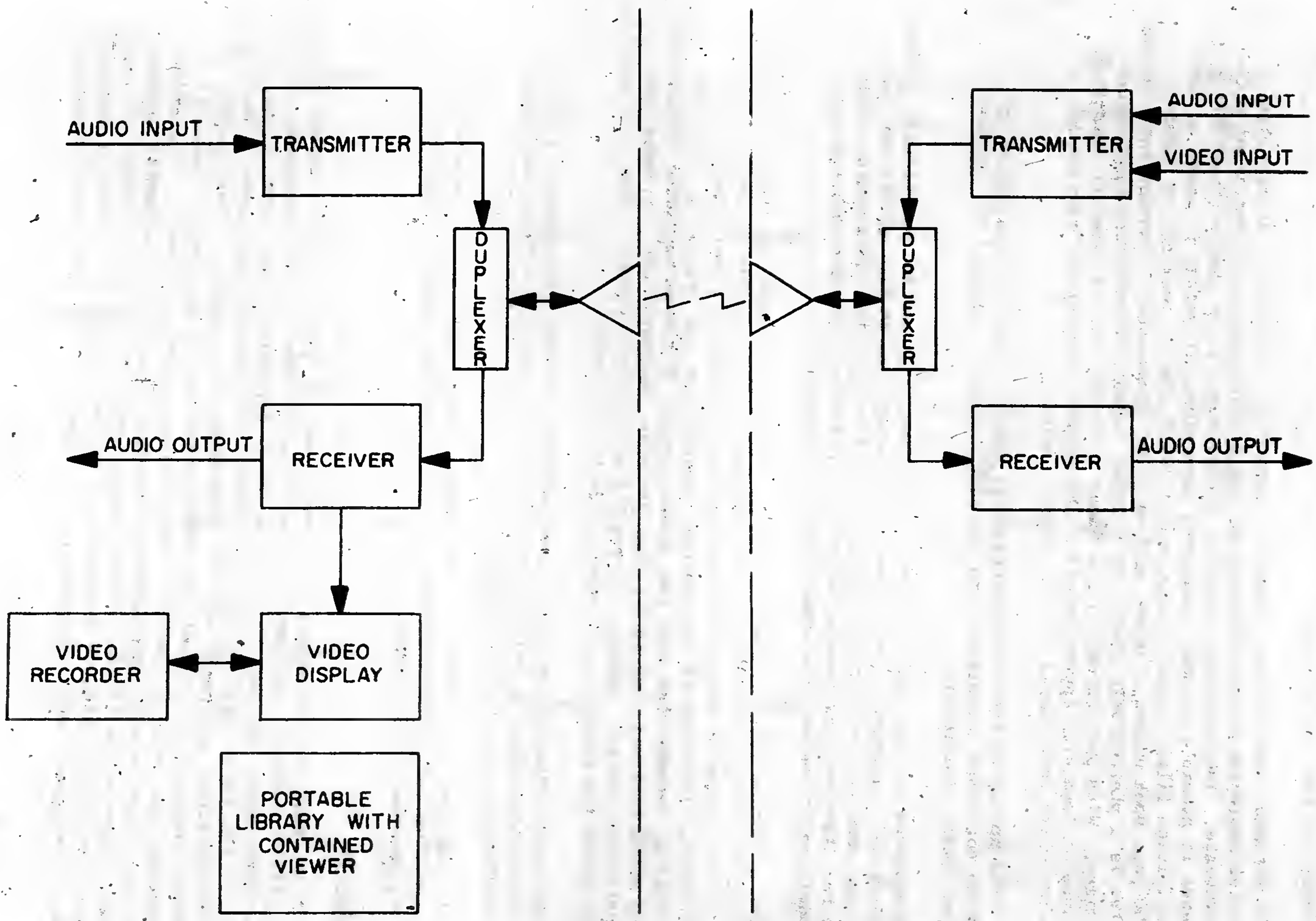

FIGURE 3 — BLOCK DIAGRAM FOR A SYSTEM WHICH CONTAINS AN ON BOARD LIBRARY

The frequency, power output, receiver sensitivity, and other RF characteristics of the communications system would be determined after a careful study was made of the flight plan, time duration, size of required library, etc., for the manned vehicle or station.

It should be noted that most of the items to be employed in this analysis are not new items that are pushing the "state of the art" but are almost all available off-the-shelf.

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NON-CONVENTIONAL COMMUNICATION

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Summary

To try to overcome the problem of time delay in radio communication with deep space missions, a possible new type of direct communication between two persons has been investigated. The method consists in stimulating the first person by his performing mental arithmetic operations or remembering names. Coinciding responses from the second person have been measured with an electronic plethysmograph or finger volume change recorder. The response signals seem to be larger for stimulation than for control periods over distances of hundreds of miles. Thus a binary code communication device of large-small signals (like dots and dashes of the ternary Morse Code) is being attempted.

Introduction

The problem of time delay in deep space radio communication is sufficiently serious to warrant searching for a solution in any possible area no matter how unconventional. If one were to be found, secondary advantages might be obtained in a system (a) less susceptible of being jammed than radio, (b) not affected by solar storms, etc. Doubtless as experience suggests, it would have its own disadvantages also. Wherever possible, conventional solutions should first be used, e.g., there is a 30-minute behind-the-moon radio communication blackout. This problem could best be solved by radio. One would need one or two reflecting moon satellites in an orbital plane at right angles to the earth-moon axis; if these are too difficult to launch from the Apollo moon craft travelling in the plane of the axis, a series of reflecting satellites with retro-rockets, released successively after the Apollo has entered moon orbit in the plane of the earth-moon axis, would suffice to overcome the behind-the-moon blackout.

The problem of time delay, though, cannot be overcome by radio. It involves a round trip time of 1 hour 20 minutes to Jupiter, 2 3/4 hours to Saturn, eleven hours to Pluto, 9 years to Alpha Centauri the nearest star. To solve this problem requires the investigation of some other form than sound or electromagnetic radiation phenomena. If any other method exists, it will be necessary to check if this also is subject to time delay. This paper is a report of four year's work investigating a new form.

Method

We are trying to develop an observation of S. Figar[5] in 1958. It is a form of direct communication measured by a plethysmograph (from plethora -- fullness of blood in the circulation). Plethysmography is a well known method which has been used for more than a hundred years in human physiology and psychology and it provides an objective record. The method is based on measuring the change in volume correlated with changes in blood circulation of the extremities of the body[3,10] or in the whole body[4]. Diminution in volume is generally interpreted as vaso-constriction and increase as vaso-dilation. The vascular reaction, based on the autonomic reaction is not, under normal conditions, subject to voluntary control. Figar used a hand plethysmograph where the subject placed his hand in a rubber glove, enclosed by a temperature controlled water bath. Above the water the airtight container was connected by a tube to a rubber diaphragm operating a light stylus on chart paper moving at 1.5 mm. per second. When the heart pumped, the pen moved up and when the heart rested, the pen moved down, tracing out a pulsed waveform baseline of approximately 70 pulses per minute.

It has been shown[1,2] that, when a person does a short period of mental arithmetic, a vasoconstriction occurs, i.e., a reduction in volume of the hand. This appears as a large rapid deflection of the baseline downwards on the chart in Figar's method[6] of measuring direct volume changes. According to this method, a card with instructions on what numbers to multiply was placed before the subject. As soon as he began multiplying the vasoconstriction occurred. The pen slowly returned to the baseline some while after the subject gave the answer. Figar allowed five minutes for the vascular reaction to subside before deciding to give a second stimulus.

In the original work Figar noticed a curious phenomenon with several subjects. As soon as he thought of picking up the card on which were written instructions about the mental arithmetic but before actually doing so, his thought was followed by a rapid vasoconstriction in his subject's hand. It seemed as if there was some kind of communication between Figar's thoughts, or emotions dependent on his thoughts, and his subject's vasoconstrictions. He went to the trouble of designing and executing an experiment[5] to measure simultaneous vasoconstrictions in two persons, one of whom (the agent) performed mental arithmetic and the other (the subject) did not know when the first did so. He used two mechanical apparatuses, deliberately rejecting electric or electronic ones to exclude any possibility of reciprocal electrical or magnetic influence. Via the tubes and diaphragms each gave a systolic-diastolic waveform on the same chart paper.

Examples of simultaneous vasoconstrictions are shown in Figures 1 and 2. In some there is an initial rise or vasodilation before the rapid vasoconstriction. In the last no agent curve was taken, but the subject produced three out of three coinciding vasoconstrictions.

The questions raised by Figar's work are whether all sensory and subliminal stimulation was eliminated so that one could say the plethysmograph vasoconstriction was a response to a mental stimulus from another person. In our Psi Communications project at Newark College of Engineering, we are trying to answer that question. If the answer is negative, then we are dealing with conventional communication subject to time delay in deep space.

Revised Method

Figar came to New York for three weeks in 1960 and taught me how to build a mechanical apparatus. I was pretty skeptical that I could get simultaneous vasoconstrictions but was surprised when they showed on the charts. I soon changed to an electronic apparatus because with the mechanical one the thin rubber diaphragm broke too easily in the middle of an experiment and the pens on the drinking straws were troublesome. The finger plethysmograph is made by the Decker Corporation of Philadelphia. It is the Decker Cardiodynameter 307-1 sensing unit and meter unit, the output signals of which feed a Massa 2-channel Meterite Model BSA-250 rectilinear electric writing recorder.

The finger plethysmograph is just as good as the hand. It will work on any finger, thumb or even the toes. A plastic cup fits over the finger on the fleshy part between the first and second knuckle. Different sizes are available to make the fit tight enough to prevent leakage but not too tight to affect the circulation. A plastic tube 11 mm. outside diameter, 8 mm. inside diameter, and 40 cm. long joined the cup to the transducer unit. This was placed in the next room with the tube passed through a small hole in the solid wall. Transmission of pulse volume changes along 3 meters was found possible using 3 mm. I.D. flattened plastic tube and metallic step down connectors. The transducer unit's circuit is based on a design by K. S. Lion[9]. In it a pressure capacitance diaphragm pickup along with the T-42 ionization transducer tube convert minute volume pulsations into large analogous electrical signals of the order of volts. Thus large amplification with its resultant noise is not necessary for obtaining a continuous record of the volume changes of the finger. The vasoconstriction shows as a large signal or D.C. baseline shift of the pulsed waveform.

Only one such electronic apparatus was used attached to the subject who lay on a bed, with his finger on the level of his heart. A marker pen indicated the time during which the agent was stimulated in the next room. We eliminated mental arithmetic as it was never possible to be sure that the agent did the mental arithmetic as requested. Instead we have used names written on cards since the agent is definitely stimulated when he reads the name. Further, with names, the stimulation can be classified as 1) names known to the subject, 2) names known to the agent, 3) phone book names as a control not known to either of them, and 4) blanks as a second control. Five of each kind have been used making twenty stimulation periods (each twenty seconds long) for one experimental session. The order of names and blanks was randomized and the time interval between stimulations was also randomly varied in a manner not known to the subject. Thus with the blank control period we could sample the physiological background deflections occurring during the whole length of the experiment to give an average natural movement for that person on that day at that time. Any larger vasoconstrictions occurring over and above the control average that might be related to the stimulus could easily be measured.

Three measurements are made as in Figure 3: 1) the vertical rectilinear baseline shift of a vasoconstriction in mm. measured from the bottom of the pulse wave in both cases, called a DIP. Dips less than one tenth of the scale are ignored. 2) the horizontal time taken for a dip to occur in seconds, named TIME, 3) the horizontal delay in seconds from the time when the card is picked up, i.e., the start of stimulus period, to the start of the deflection downwards, called the LAG.

The measurements have been made on a double blind basis. Only dips occurring during a period are measured. The periods are numbered 1 through 20, but otherwise the measurer does not know which is which. Most of the measurements have been done this way by Dr. Carroll B. Nash of St. Joseph's College, Philadelphia, Pa.

The test then of whether a new form of communication exists rests on whether on average larger deflections occur during name periods than during blanks, on a reproducible statistical basis. If it does, this presents us with a method for converting into a communication system since the large deflections and small deflections can act as the dots and dashes of the Morse Code.

Results

After the subject lay down, his finger was connected to the instrument. His elbow and also his wrist were supported by foam rubber so he could keep them still for about twenty minutes. The system was tested for leaks and the pen writer was calibrated. About 15 minutes was allowed to elapse while temperature equilibrium of the finger (at 98.6° F) and apparatus (at room temperature of 75° F ± 4° F) was established. The subject also became relaxed and less likely to move during the experiment.

The names asked for were of persons met recently who had caused considerable emotional impact. The randomizing was done in blocks of four to prevent five of one kind coming together and coinciding with a series of vasoconstrictions by an artifact.

The agent, with a solid wall between him and the subject, sat at a desk. When the chart tracing was steady and centered, he waited for the sweep second hand of a watch to come to the 12 position, picked up the first card, and turned on the marker pen. He looked at the name or blank for twenty seconds and then placed it face down and turned off the marker switch. After waiting for the required random interval as on the prepared time schedule he picked up the second card and so on to the last card.

An example of the kind of curve obtained is shown in Figure 4. The subject's name was Mr. Estebany from Hungary, and I did not know him before the experimental series started. The chart time goes from left to right. The left point of the marker pen corresponds to the start of agent stimulation where he picked up the card and read the name. Twenty seconds later, the right point is where the card was put down or the end of stimulation.

#6 is an agent name and there is no plethysmographic response. #7 is a blank and no response. #8 is the name of the subject's sister, and there is a large vasoconstriction the start of which coincides with the start of the agent's stimulation. She lived in Hungary and normally he was afraid to write to her. However, a family crisis came up and he had to write, the evening before this curve was taken. This suggested that recency and strong emotionality were possible factors.

The statistical results of this first New York series are shown in Table 1. Reading the DIP line, it is seen that during 55 blank periods the average deflection measured 9.00 mm. During the phone book control and agent name periods, the average dip was 8.40 and 9.65 mm. These three means are not different within the experimental error. The subject and subject-agent names averaged 14.16 and 14.45 mm. which are also not sensibly different from each other but they are significantly larger (1.5:1) than the blank mean. A t test gives a probability P beyond the 1% level using the two-tailed test. Reading the TIME line, it is seen that the TIME measurements back up this finding. (The DIP and TIME measurements are correlated with a Pearson r = 0.72). From the LAG line one can see that the blanks averaged 8.97 seconds where 10.00 seconds is expected on a random basis; but that the subject's names are associated with a shorter delay but not significantly so.

With regard to the possibility of error in measuring the curves, an error of 2.5-3 mm. by the judge acting positively for all the blank control values and negatively for all the subject name values is required to close the difference between these means. It should be stressed that the measurement was done blind without the person knowing which period was a stimulation name and which a blank control. Each reading was checked independently by a second judge and any differences agreed upon. After this the code of the order and the measurements were exchanged. At this point the double blind character is lost but then no measurements can be changed.

Later I moved to Newark College of Engineering, New Jersey. It was of interest to see if I could reproduce my own results in a new place with new subjects. The diagram in Figure 5 shows the two adjacent rooms we used. Figure 6 shows a part of a curve obtained. Stimulus #1 is a phone book control with no change in the baseline pattern. Stimulus #2 was a name and the response occurred very accurately synchronized with the start of the stimulus. Figure 7 shows stimulus #17, a name appearing with a large response, and #18 a blank stimulus and no response. Figure 8 shows part of a curve given by Professor Mihalasky who teaches Industrial and Management Engineering. Stimulus #1 was a blank, no response; #2 an agent name with a small response; and #3 the name of the subject's two-day old new-born baby: this name coincided with a large vasoconstriction which took over a minute to get back to the original baseline level. Again the factor of recency and strong emotionality seem to be important.

Figure 9 is part of a curve from a student, Mr. T. Kesolits. He was conditioned by electric shocking to sets of three call letters of an amateur station written on a card instead of providing names of persons he knew. The shocking lasted 20 seconds each for eight sets. There were seven control sets without shocking and five blank controls which were viewed randomly by the agent in the usual way. Stimulus #14 is a blank with no response. #15 no shock with a small reaction, and #16 a shocked stimulus with a large vasoconstriction occurring.

Figure 10 shows another student's reaction from Mr. D. Fenily. This was done over a distance of 35 yards with no marker pen. Instead the time periods were added in afterwards by measuring the time from a starting point. Stimulus #3 is a neutral phone book name - no reaction; #4 is an agent name - a large accurately timed vasoconstriction, while the blank #5 occurs during absence of response.

It is understood that all the examples shown could have occurred by chance so let us pass now to the statistical results. Table 2 shows that from December to

March 1963 in experiments A 21-30 with Professor Mihalasky 50 subject name stimuli occurred at the same time as DIPS averaging 9.62 mm., contrasted with the mean of 50 blank periods of 3.46 mm., nearly a 3:1 ratio. Again the TIME values back up this finding; the LAG also is less for subject names than blanks which average close to the expected 10 seconds. In the next Table 3, a fresh random control order was applied and the means were recalculated. The ratio was reduced to 1:1 as expected. Part of the reason for the larger ratio of 3:1 than the original 1.5:1 was due to an advance in measurement. In this we included only rapid vasoconstrictions; under standard conditions the DIP had to be equal to or greater than the TIME, or it was ignored. It is interesting to compare the actual measurements of the subject name reactions with those of the blanks as in Table 4. There are 23 non-zero subjects to only 11 blanks, i.e., twice as many and they are larger. A 2 x 2 chi square test of zero-non zero shows this to be a difference significant at the 1% level. Thus it seems that the original results were able to be repeated in a new place.

In the next series, A 31-40, we flashed a light to let Professor Mihalasky know consciously when the stimulus was given but not which one it was. The results are less with a DIP ratio of 1.7:1. He stated that far from helping, the light hindered by interrupting his stream of thinking. Table 5 shows both sets of results; December-March blank average was 3.46 mm. and March-May was 3.57 mm. suggesting a stable control. Here the agent mean dropped from 8.04 to 3.59 mm. close to the blank control. In the A 21-30 series the agent gave names of persons met in the last few days with considerable emotional impact. In the second they were old names with no emotionality, suggesting again that recency and emotionality are factors.

In the experiments using adjacent rooms, it was possible that when the agent read the name he subliminally voiced it and that sound communication travelled throug the hole from the tube to the subject. So we gave up using adjacent rooms and went on to distance experiments. These were done between two buildings more than one-eight of a mile apart. Experiments A 41-50 with Professor Mihalasky show a ratio of 2.0:1 as in Table 6. Experiments W 01-10 with Mr. D. Fenily, a student, over 75 yards show a ratio of 1.7:1. However, with Fenily the agent names are associated with deflections much bigger than the blanks and the subject names much less so. This seems to be a reproducible effect in experiments with him.

At this point a double copper walled screened room was placed at our disposal by the Electrical Engineering Department of the College. This has a door with inside and outside flanges of copper that press against the metal covered wall. When the door is open, the strongest radio station, WOR, in Newark, New Jersey, can be heard very loud at full volume. With the door closed, the reception disappeared completely. Testing with a noise meter showed that radio attenuation over the medium waveband was 95%.

Table 6 shows the mean plethysmograph results obtained with Professor Mihalasky and Mr. Fenily when the agent was inside the screened room and all randomizing of order and timing of stimuli was done after the double copper door was closed. The results seem to be similar with the agent both inside and outside the screened room. Again, for Professor Mihalasky the high ratio is for subject names and for Mr. Fenily it is for agent names.

It is understood that the results must be repeated with a 99.99% attenuation room over the widest possible band, which the College is building. However, the plethysmograph method seems to lend itself to such experimental design. If the results continue, then we will have reached the point Vasiliev[12] in Leningrad reached in 1934 on long distance and in 1936 on screening. His method of direct communication was the use of hypnosis to suggest sleep and waking. The subject 1100 miles away seemed to obey. The results were also positive when the subject was inside a screened lead chamber sealed with mercury. If the plethysmograph and hypnosis methods of direct communication are similar, then they seem to have different properties from radio with regard to screening.

Communication System

If the plethysmograph reaction is a reproducible effect, how is it proposed to develop a communications device? The design in its engineering essentials has already been worked out by R. Taetzsch[11]. The design is based on sequential sampling plans as used in quality control in industry. One continues to transmit information until a decision is reached thus increasing the redundancy in order to increase the reliability.

To transmit a message, we would switch over from 1) random times to fixed time slots (say every minute) and from 2) random stimulus orders (name or blank) to orders based on the message in some binary code. A name stimulus would be used in the first slot if the message translated as in a dot of the Morse Code first; conversely a blank would be used if the first one was a dash. At the receiving end a large plethysmograph deflection in the first time slot would be translated as a dot and a small deflection as a dash. Each letter of the alphabet could be coded over some number of time slots dependent on the reliability required.

The process is slow, but this is the price paid for reliability. We estimate that the above results are equivalent to about one bit per five minutes with two out of three reliability. Speeding up may come by connecting directly into the sympathetic nervous system rather than depending on the ensuing vasoconstriction of the peripheral blood vessels, and the slow return to base level.

The process can also go into reverse. This would occur if the blank deflections were larger than those corresponding to the name stimulus. We have observed this and noticed it is very evident with students at the time of a tough examination or when they receive the news of flunking a test. Since we are measuring unconscious mental reactions, it is to be expected, just as muscles expand and contract, conditioned reflexes inhibit and disinhibit, etc. However, Taetzsch's system provides a means of dealing with this problem. At intervals which are known to transmitter and receiver, sample stimuli also known to transmitter and receiver are transmitted. If they are received as sent, the unknown received message is accepted; if they are received reversed, the received message is then reversed so as to duplicate the transmitted message.

In conclusion, the first part of trying to solve the time delay problem has been to look for some other form of communication. This is perhaps being like back in the days of Hertz in 1886 when Hertzian waves were being investigated long before crystal detectors, vacuum tubes, feedback oscillators, and transistors were known to amplify the effect. The second part, if some other form seems to be found, is to determine if the new form has some properties different from those of radio as perhaps occurs in screening experiments. If this is so, it may be that the non-conventional communication has different time delay properties also. The third part then is to try it out in space to see what kind of time delay the new form exhibits.

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Table 1.

Mean Plethysmograph Responses

Mr. Estebany New York, February 1961 Temperature 23°C ± 2°C

Response Measured	Stimulus Name Known to:				
	Subject Agent SA	Subject Only S	Agent Only A	Neither Control N	Blank Control B
No. Stimulus Periods	33	44	49	24	55
Mean DIP mm.	14.45	14.16	9.65	8.40	9.00
t ref. B	2.96	2.55	0.32	-0.27	-
P	0.004	0.012	n.s.	n.s.	-
Mean TIME Sec.	9.62	9.14	6.82	7.15	6.46
t ref. B	2.87	2.42	0.22	9.33	-
P	0.005	0.017	n.s.	n.s.	-
Mean LAG sec.	7.79	8.07	8.03	6.47	8.97
t ref. B	-0.87	-0.83	-0.81	-1.68	-
P	0.39	0.41	0.42	0.10	-

The t values were obtained using Fisher's formula for the significance of the uncorrelated means. The t test in the table represents the difference of the means in the first four categories from the blank control means.

Table 2

Mean Plethysmograph Responses

Professor J. M.   A 21-30, December 1962 - March 1963   Temperature 23° C ± 2°C

Response Measured	Stimulus Name Known to:			
	Subject only S	Agent only A	Phone Book Control N	Blank Control B
No. of Stimulus Periods	50	50	50	50
Mean DIP mm	9.62	8.04	7.56	3.46
Mean TIME sec.	1.58	1.16	1.22	0.575
Mean LAG sec.	7.59	8.2	8.2	9.82

Table 3

Mean Plethysmograph Responses

A 21-30 Professor Mihalasky Dec. '62 - Mar. '63 Temperature 23°C ± 2°C

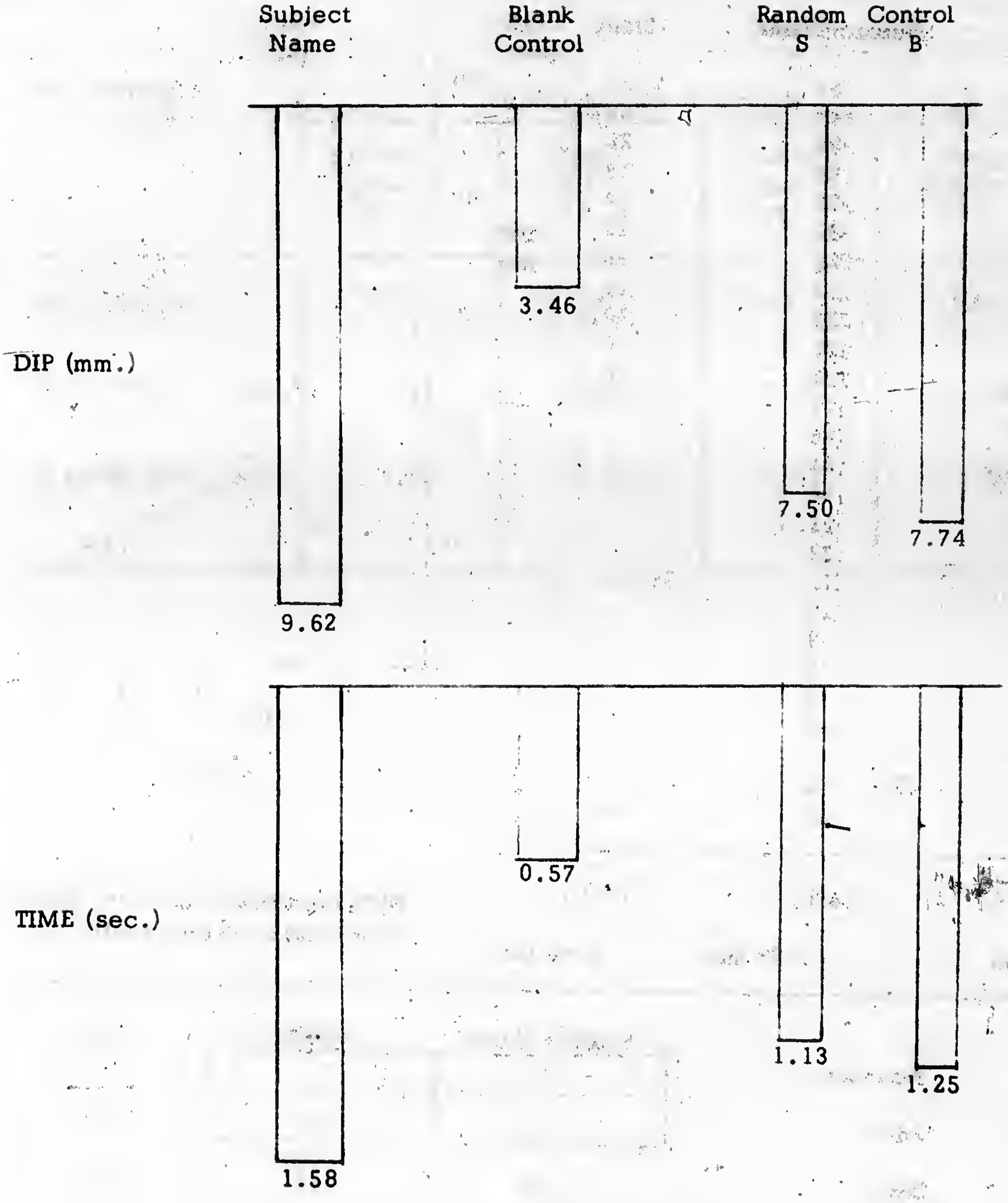
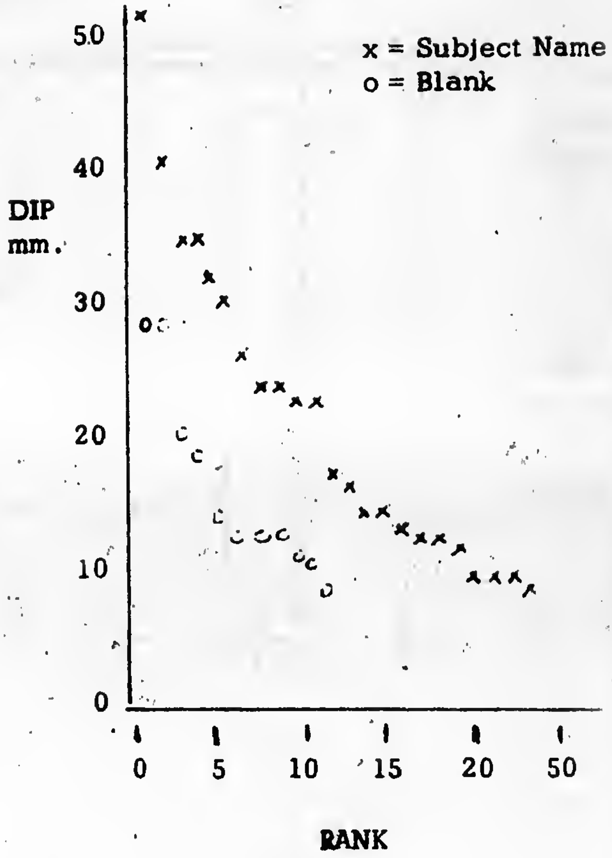


Table 4

Professor J. M., Newark A 21-30

Ranking of DIPS During Subject Name-Blank Periods

Rank	Subject Name	Blank
1	51	28
2	40	28
3	34	20
4	34	18.4
5	32	14
6	30	12
7	26	12
8	23	12
9	23	10.6
10	22	10
11	22	8
12	17	0
13	16	
14	14	
15	14	
16	12.6	
17	12	
18	12	39 x 0
19	11	
20	9	
21	.9	
22	9	
23	8	
24	0	
.	27 x 0	
50	0	0
	480.6	173.0
Mean	9.62 mm.	3.46 mm.



DIPS less than 8 mm. are ignored as less than 1/10th scale.

	Subject Name	Blank	Total
Non-zero	23	11	34
Zero	27	39	66
Total	50	50	100

$X^2 = 6.42$ (1 d.f.) P = .01

Table 5

Mean Plethysmograph Responses

DIP Measurements in mm. Prof. Mihalasky Temp. 23° C ± 2° C

	Stimulus Name Known to:			
	Subject only S	Agent only A	Phone Book N	Blank Control B
No. Stimuli	50	50	50	50
A 21-30 (Dec.-Mar.)	9.62	8.04	7.56	3.46
A 31-40 (Mar.-May)	6.08	3.59	6.89	3.57

Table 6

Mean Plethysmograph Responses

DIPS (mm.) Newark Unchecked Temperature 23° C ± 2° C

	NO SCREEN				DOUBLE COPPER SCREEN		
	Stimulus Name Known to				Stimulus Name Known to		
	Subject only	Agent only	Blank control		Subject only	Agent only	Blank Control
Prof. M. A 41-50 Dec.- Mar.'63 250 yds	27.1	27.2	13.5	Prof. M. A 51-58 Dec.- Mar.'64 35 yds	18.0	15.8	10.6
D.Fenily W 01-10 Oct.- Dec.'63 75 yds	14.2	20.1	12.6	D.Fenily W 11-18 Dec.- Mar.'64 35 yds	15.4	18.0	11.5

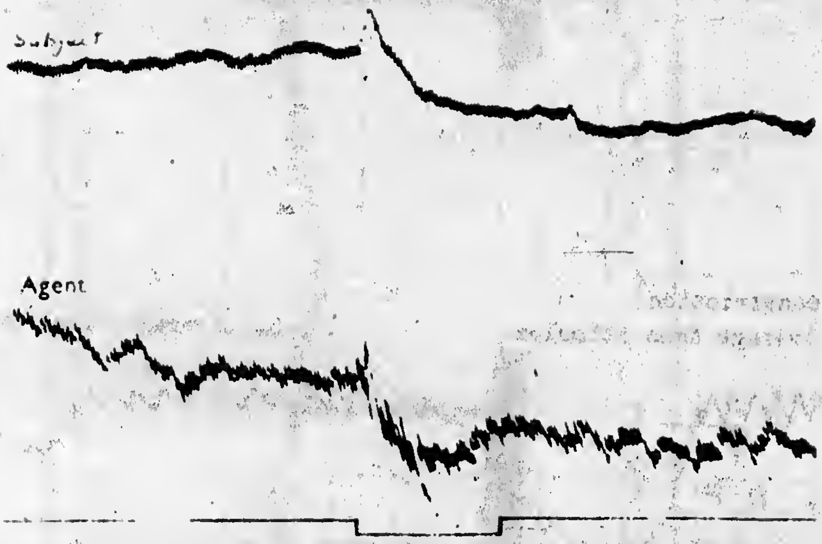


Fig. 1
Figar's Coinciding Vasoconstrictions.

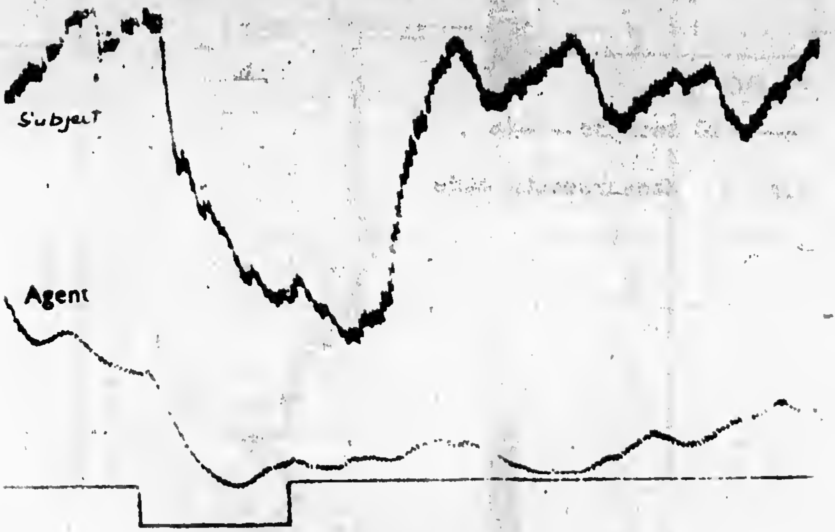


Fig. 2
Figar's Coinciding Vasoconstrictions.

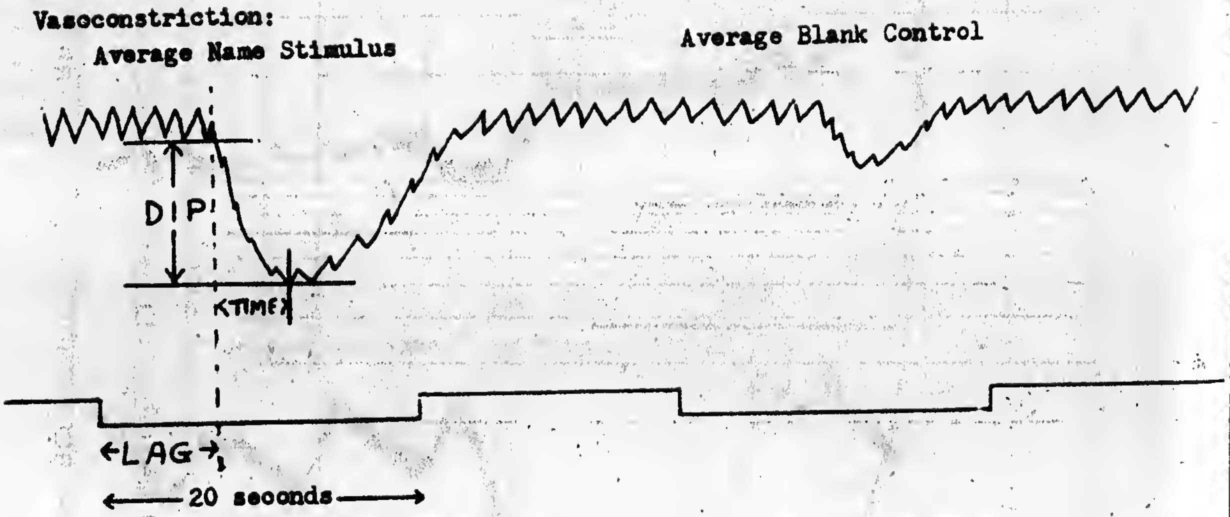
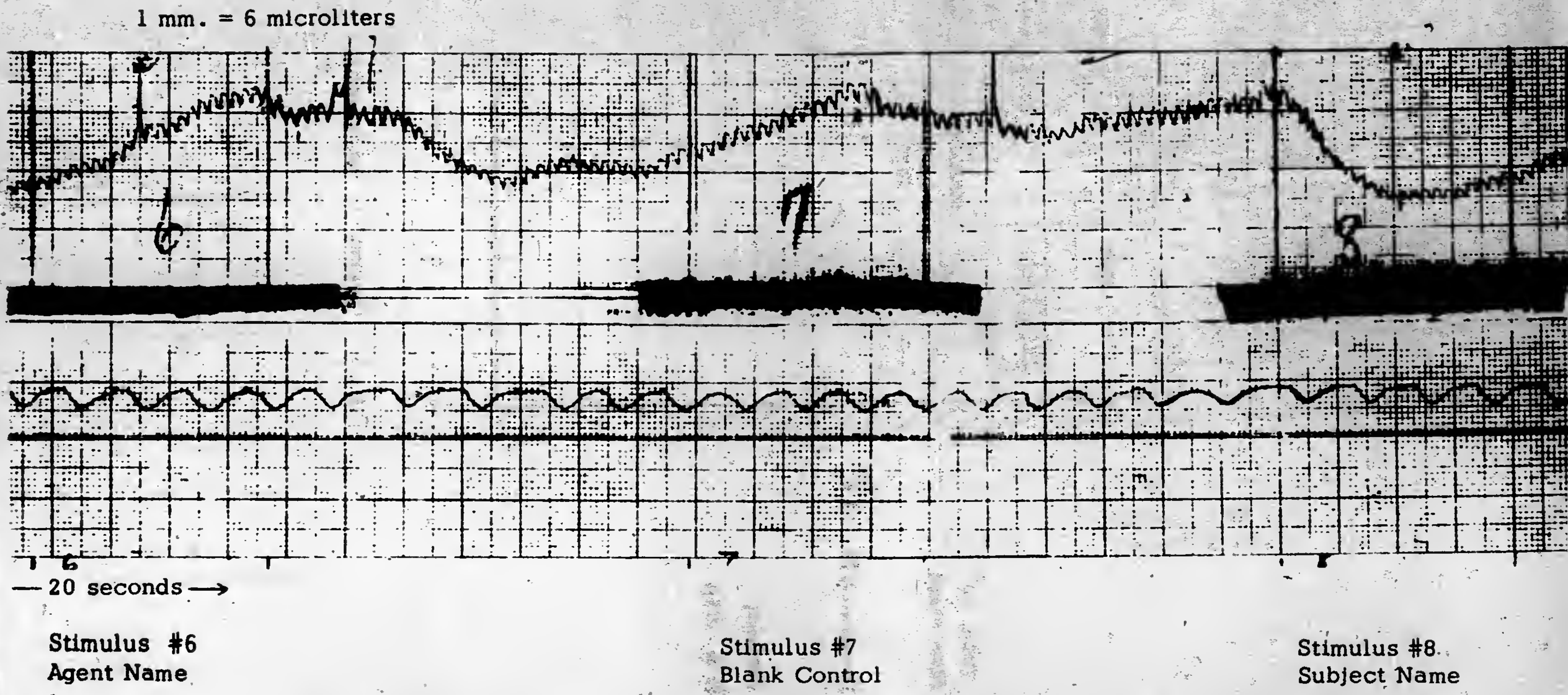


Fig. 3 Measurements made.

Fig. 4 Subject Mr. Estebany. In the double blind procedure the marker pen records are covered up with masking tape to prevent the person making the measurements from knowing that 7 was unmarked and a blank control.



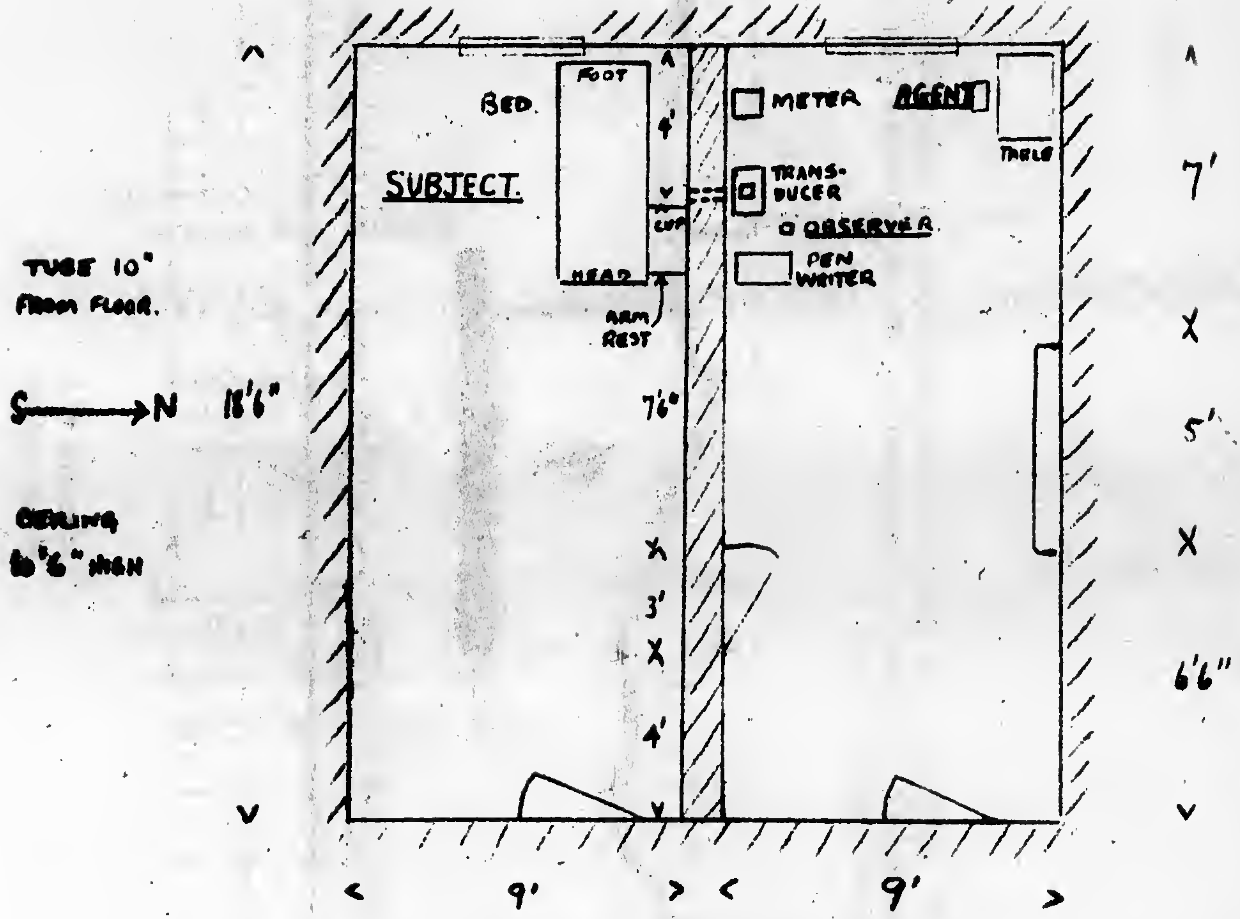
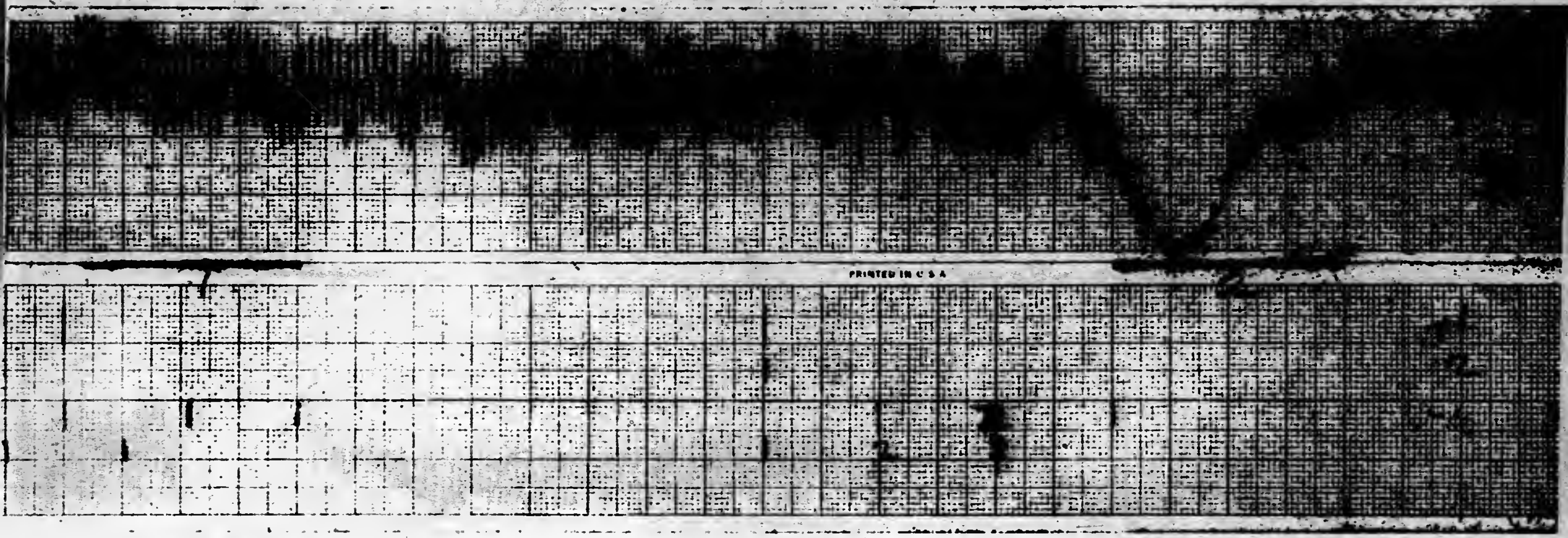


Fig. 5 Diagram of Experimental Situation.

1 mm. = 1 microliter. Fig. 6 Mr. C. Experiment 0 01.

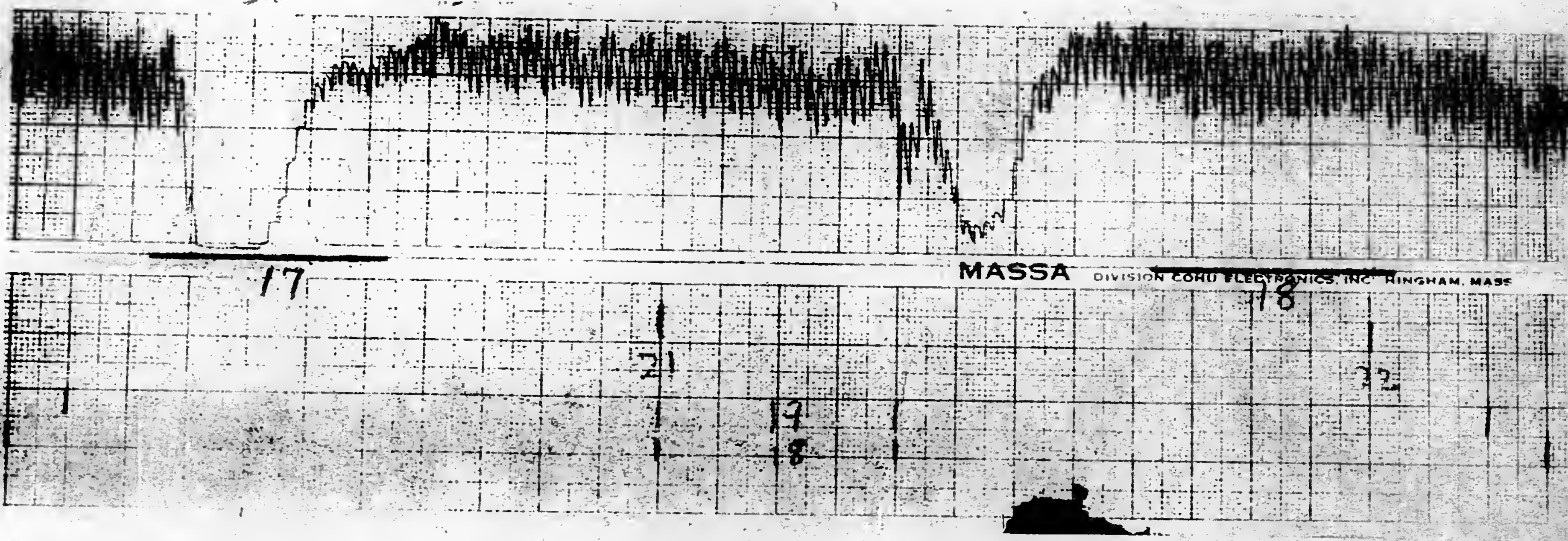


1 mm. = 1 microliter

Fig. 7

Mr. C.

Experiment 0 01

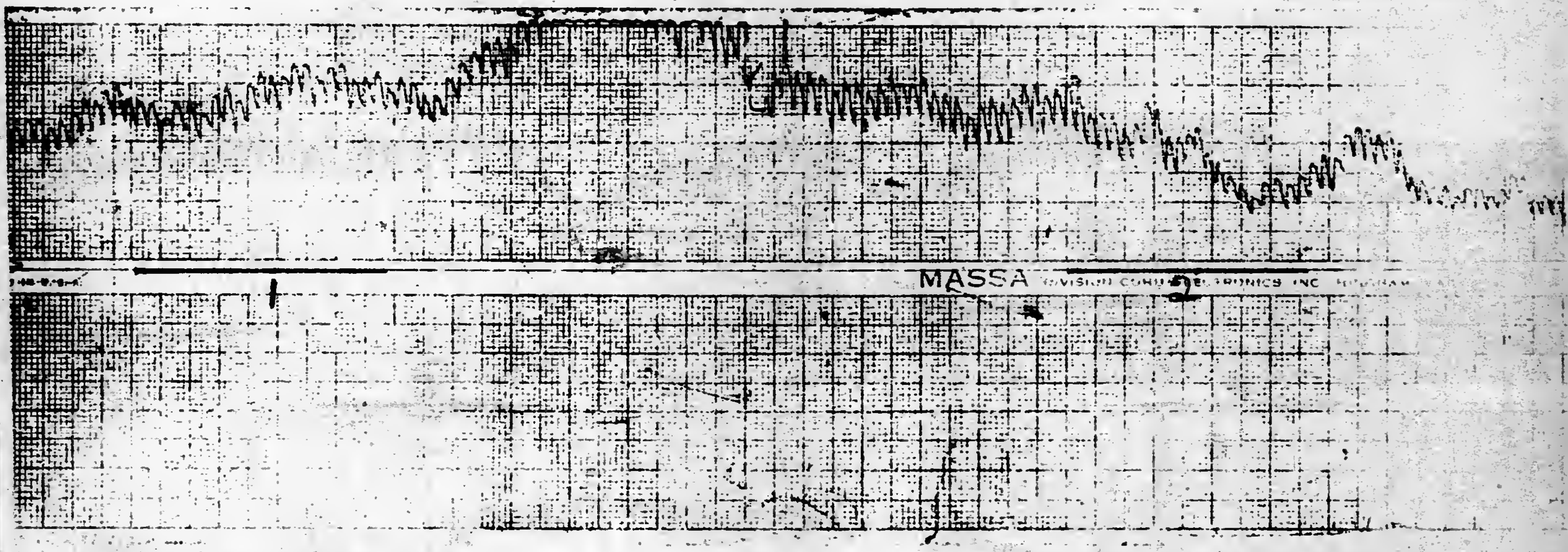


← 20 seconds →

Stimulus #17
Subject Name

Stimulus #18
Blank Control

1 mm. = 2 microliters Fig. 8a Prof. M. Experiment A 11

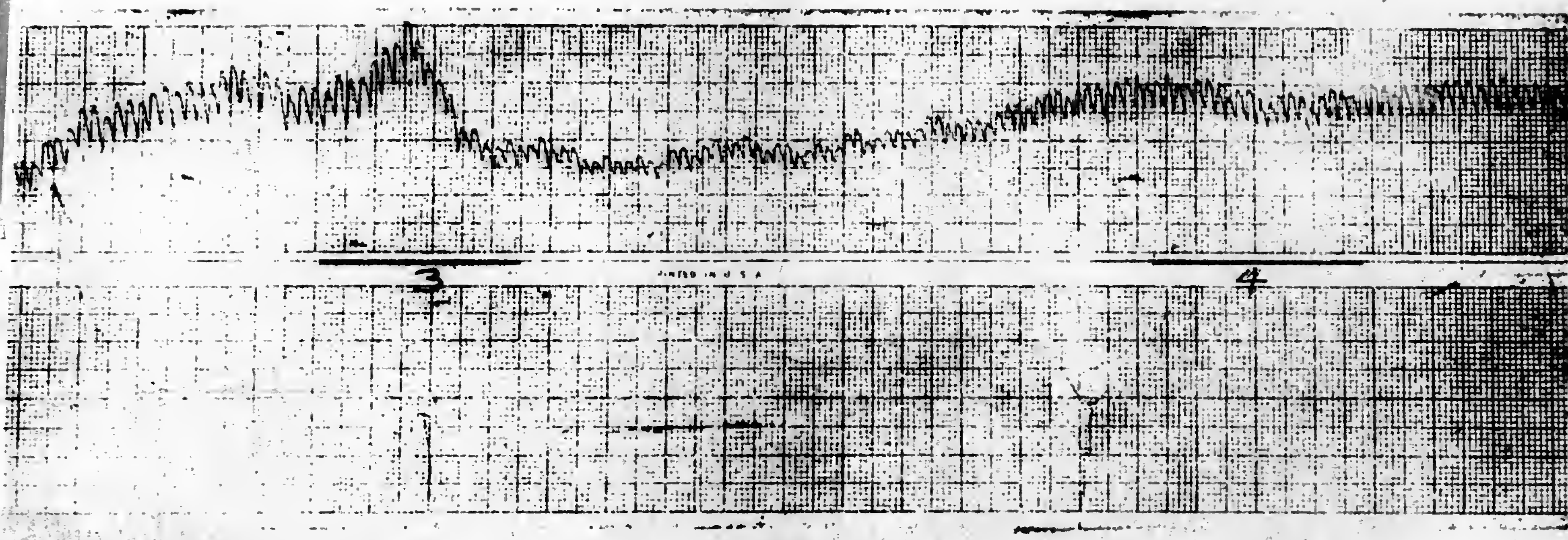


←20 seconds →

Stimulus #1
Blank Control

Stimulus #2
Agent Name

Fig. 8b



Stimulus #3
Subject name
Newborn baby

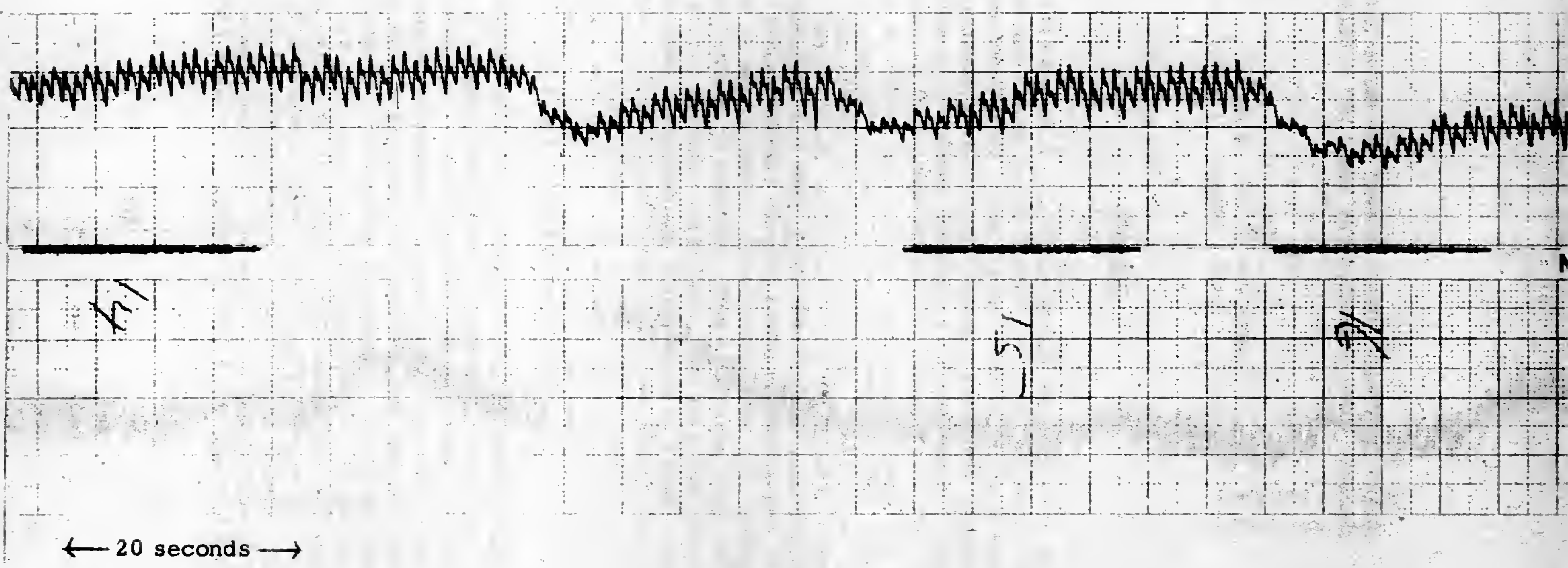
Stimulus #4
Phone [illegible]

1 mm. = 5 microliters

Fig. 9

T. Kesolits
Electric Shock Conditioning

Experiment RS 09

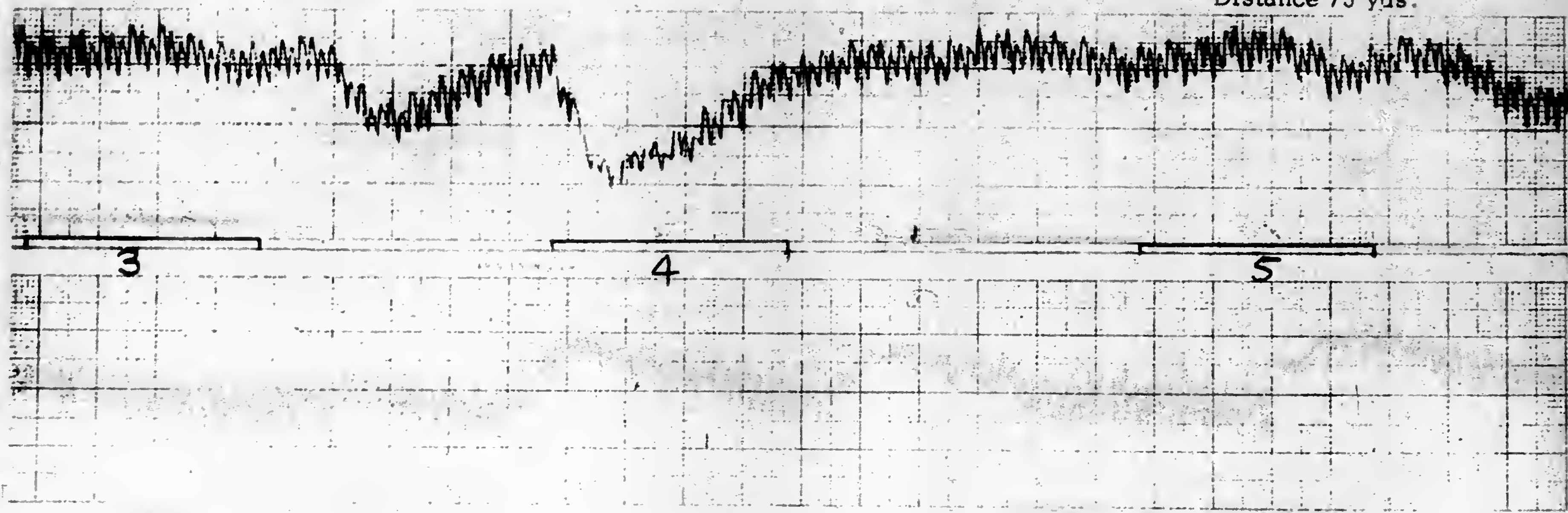


Stimulus #14
Blank Control

Stimulus #15
No Shock Letters

Stimulus #16
Shocked Letters

1 mm = 2 microliters Fig. 10 D. Fenily Experiment W 07
Distance 75 yds.



20 seconds

Stimulus #3
Phone book control

Stimulus #4
Agent Name

Stimulus #5
Blank Control

THRUST MOUNT AND LAUNCH STAND STRUCTURAL DESIGN CRITERIA FOR LARGE MISSILES AND SPACE VEHICLES

By

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SUMMARY

Often it is necessary to begin design studies of a missile launch stand and thrust mount before the final missile configuration has been determined. The thrust mount structure should provide: (1) dynamic stability for the missile-stand combination during captive and holddown firings, (2) sufficient stiffness to prevent excessive displacements during engine start and shutdown, (3) adequate stiffness for hot and cold engine gimbaling checkouts, and (4) sufficient stiffness to provide safe dynamic loads that may result from engine firing malfunctions. This paper provides generalized basic design criteria for the stiffness or spring constant requirements that should exist at the vehicle attachment points for booster thrusts ranging between 300,000 and 30,000,000 lb. Failure to meet these criteria could result in damage to a costly space vehicle or missile. These criteria also include overall vertical, lateral, rotational, and torsional stiffness requirements that are presented graphically as thrust versus stiffness. From a plot of launch stand stiffness versus vehicle-stand frequency, it is shown that the fundamental frequency increases with increasing stiffness up to a definite value of stand stiffness. Thereafter, almost no frequency increase is gained by strengthening the stand beyond this stiffness value.

The method of determining launch stand stiffness requirements is based on a dimensionless matric formulation involving free flexural vibrations of missiles mounted on a flexible stand. The effects of variable boundary conditions at the base support, variable bending stiffness along the vehicle, and variable mass distribution along the vehicle are considered. A thrust-to-weight ratio of 1.3 is used since it was found that a ±15 percent variance of this ratio did not appreciably change the stiffness requirements of the launch stand. Bending stiffness at the base of the vehicle is established as a function of booster thrust. Also, the average length of the vehicle as a function of thrust is established based on existing missiles and space vehicles.

Any number of lumped masses along the vehicle can be used. For a parameter study of this nature, ten masses are chosen to ensure sufficient accuracy. The problem is matrically formulated in standard eigenvalue form for rapid solution on an electronic digital computer.

The criteria presented here are approximate and cannot apply to all launch stands exactly; however, the criteria requirements are reasonably accurate and should permit design of a particular launch stand to begin early in the program. For example, these criteria were used in establishing the Saturn V launch stand stiffness requirements at the request of NASA early in 1963. Similar criteria could now be established for a Post-Saturn launch stand or any other large missile or space vehicle launch facility.

NOTATION

a_i	Coefficient relating flexural stiffness at i^{th} station to flexural stiffness at the first station
b_i	Coefficient relating mass at i^{th} station to mass at the first station
E	Modulus of elasticity in lb/in.^{2}
f	Natural frequency in cps
h	Spacing between stations along vehicle length in inches
i	Subscript which refers to parameters at a particular station
I_i	Average area moment of inertia of vehicle cross section at i^{th} station in inch^4
k_i	Longitudinal spring constant between i^{th} and i^{th-1} stations along vehicle length in lb/in.
k_h	Lateral spring constant at base of vehicle in lb/in.
k_v	Vertical spring constant at base of vehicle in lb/in.
L	Total length of idealized vehicle in inches
m_i	Mass at i^{th} station in lb-sec^2/in.
q	Last or highest numbered station
r_i	Radius of idealized vehicle at i^{th} station in inches
R	Rotational spring constant at base of vehicle in in.-lb/rad
R_T	Torsional spring constant at base of vehicle in in.-lb/rad
T	Booster thrust of idealized vehicle in pounds
u_1	Boundary condition parameter
u_2	Boundary condition parameter
W	Weight of idealized vehicle in pounds

y_i	Lateral deflection at i^{th} station
ϵ	Dimensionless boundary condition parameter
λ	Dimensionless natural frequency parameter
ω	Circular frequency

I. TECHNICAL APPROACH

The method of determining launch stand stiffness requirements is based primarily on a dimensionless matric formulation given in Reference (1). Total stiffness requirements will be developed for the lateral, rotational, vertical and torsional directions. Figure 1 shows the idealized vehicle attached to the launch stand springs. From Reference (1), the basic equation in matric form for the solution of eigenvalues and eigenvectors is given as:

$$[AB + \epsilon S + \zeta N - \lambda(HM - \alpha C - \beta D)]Y = 0. \; . \; . \; . \qquad [1]$$

where:

ϵ = Boundary spring parameter = $\frac{k_h h^3}{EI_1}$,

λ = Eigenvalue = $\frac{m_1 \omega^2 h^3}{EI_1}$,

ζ = Axial load parameter,

α = Shear parameter,

β = Rotary inertial parameter.

Matrices will be defined after modifying Equation [1].

For preliminary design purposes, one can neglect the effects of axial load, shear distortion and rotary inertia in the calculation of launch stand spring constants. Therefore, Equation [1] can be reduced to:

$$[AB + \epsilon S - \lambda HM]Y = 0 \; . \; . \; . \; . \; . \; . \; . \; . \; . \; . \qquad [2]$$

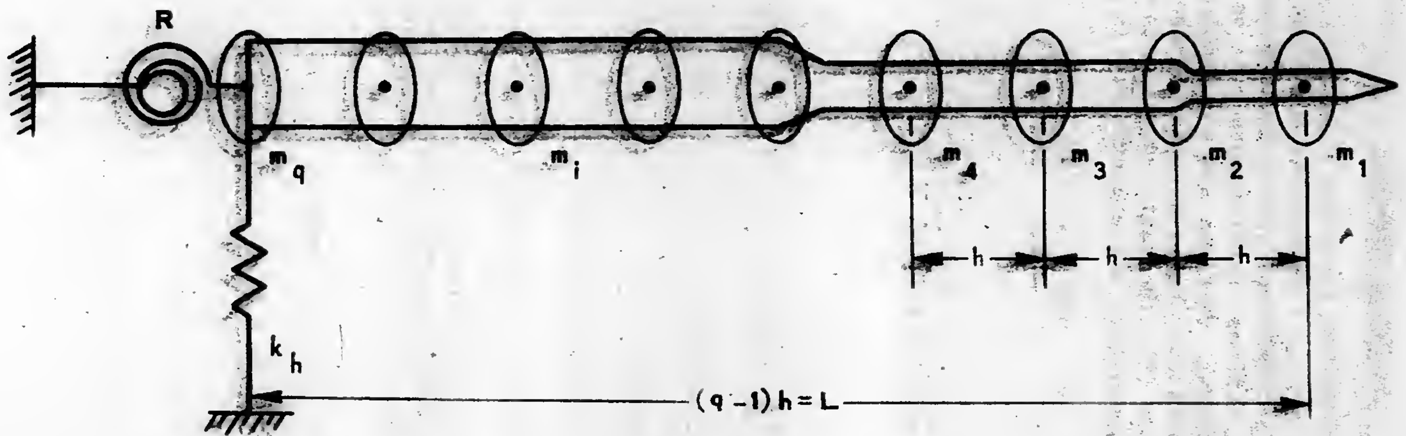

FIGURE 1 IDEALIZED MODEL FOR LATERAL AND ROTATIONAL SPRING STUDIES

$$A = \begin{bmatrix} a_2 & & & & \\ & a_3 & & & \\ & & a_4 & & \\ & & & \ddots & \\ & & & & a_q \\ & & & & 0 \end{bmatrix}_{q\mathrm{x}q-1} \qquad B = \begin{bmatrix} 1 & -2 & 1 & & & \\ & 1 & -2 & 1 & & \\ & & \ddots & \ddots & \ddots & \\ & & & 1 & -2 & 1 \\ & & & & 1+u_1 & u_2-2 \end{bmatrix}_{q-1\mathrm{x}q}$$

$$S = \begin{bmatrix} 0 & & & 0 \\ \cdot & & & \cdot \\ \cdot & & & \cdot \\ \cdot & & & \cdot \\ \cdot & & & \cdot \\ \cdot & & 0 & \cdot \\ 0 & \cdot & \cdot & 1 \end{bmatrix}_{q\mathrm{x}q} \qquad H = \begin{bmatrix} 1 & & & & & & \\ 2 & 1 & & & & & \\ 3 & 2 & 1 & & & & \\ \cdot & & & \ddots & & & \\ \cdot & & & & & & \\ q-1 & q-2 & & & \cdot & 1 & \\ 1 & 1 & 1 & 1 & 1 & 1 & 1 \end{bmatrix}$$

Equation [2] can be rewritten in standard eigenvalue form by combining matrices.

$$\text{Let } A'B' = AB + \epsilon S.$$

Now Equation [2] becomes:

$$[A'B' - \lambda HM]Y = 0 \qquad [3]$$

$$A'B' = \left[\begin{array}{c|c|c|c|c|c|c|c}
a_2 & -2a_2 & a_2 & & & & & \\ \hline
 & a_3 & -2a_3 & a_3 & & & & \\ \hline
 & & a_4 & -2a_4 & a_4 & & & \\ \hline
 & & & \cdot & \cdot & \cdot & & \\ \hline
 & & & & \cdot & \cdot & \cdot & \\ \hline
 & & & & & \cdot & \cdot & \cdot \\ \hline
 & & & & & & a_q(1+u_1) & a_q(u_2-2) \\ \hline
 & & & & & & & \epsilon
\end{array}\right]$$

u_1 and u_2 are boundary condition parameters defined in Reference (1) as:

$$u_1 = \frac{(R/2) - \left(EI_q/h\right)}{(R/2) + \left(EI_q/h\right)}, \quad u_2 = \frac{\left(2EI_q/h\right)}{(R/2) + \left(EI_q/h\right)} \quad . \; . \; . \; [4]$$

It can be shown that: $u_1 + u_2 = 1$.

All matrices of Equation [3] are in dimensionless form except for the eigenvector, Y. This feature enables one to make an extensive parameter study of a broad range of missiles and space vehicles. Average mass and bending stiffness ratio distributions must be determined along a typical multistage space vehicle. Parameters to be varied are the stand springs. The practical upper and lower limits of these springs for any given thrust can then be determined by inspecting the variance of the frequency parameter with the spring stiffness parameters. This inspection reveals that, up to a point, increasing spring stiffness increases the frequency parameter; however, beyond a certain point or range, a continued increase in spring stiffness does not appreciably increase the frequency parameter, λ.

II. DETERMINATION OF LATERAL AND ROTATIONAL STAND SPRINGS

Studies were made to determine the bending stiffness at the base of several U.S. missiles and space vehicles. Results are shown in Figure 2 which shows how EI at the vehicle base varies with thrust. The relationship between missile length and thrust is also needed, which is shown in Figure 3. Although some variation is certain to exist, these curves depict the average length and bending stiffness as a function of thrust of known missiles and space vehicles. The existing trend is then projected for thrusts up to 30,000,000 pounds.

The required values or stiffnesses of the lateral stand spring, k_h, and the rotational stand spring, R, may be determined as a function of thrust. For a solution of k_h, the rotational spring is made infinite. Similarly, in solving for R, the lateral spring is made infinite. A lumped mass system of ten masses is considered to ensure sufficient accuracy in the final results. The following mass and bending stiffness ratio distribution (Table 1) which agrees reasonably well with existing vehicles, is assumed for all booster thrusts:

Table 1 Mass and Bending Stiffness Ratios

Station	1	2	3	4	5	6	7	8	9	10
Percent of Total Mass	0.03	0.04	0.05	0.08	0.09	0.11	0.14	0.15	0.15	0.16
Percent of EI_{10}	0.125	0.1667	0.250	0.4167	0.500	0.6667	0.8333	0.9167	1.0	1.0

In the solution for lateral stiffness, k_h, a missile or space vehicle with any given thrust in the range under consideration may be used for establishing the stiffness criteria of the entire thrust range, since dimensionless parameters are used. With reference to Equation [3] and Table 1, it follows that:

$$A' = \begin{bmatrix} 1.33 &&&&&&&&& \\ & 2.00 &&&&&&&& \\ && 3.33 &&&&&&& \\ &&& 4.00 &&&&&& \\ &&&& 5.33 &&&&& \\ &&&&& 6.67 &&&& \\ &&&&&& 7.33 &&& \\ &&&&&&& 8.00 && \\ &&&&&&&& 8.00 & \\ &&&&&&&&& 1.00 \end{bmatrix}$$

since $a_2 = EI_2/EI_1$, $a_3 = EI_3/EI_1$, etc

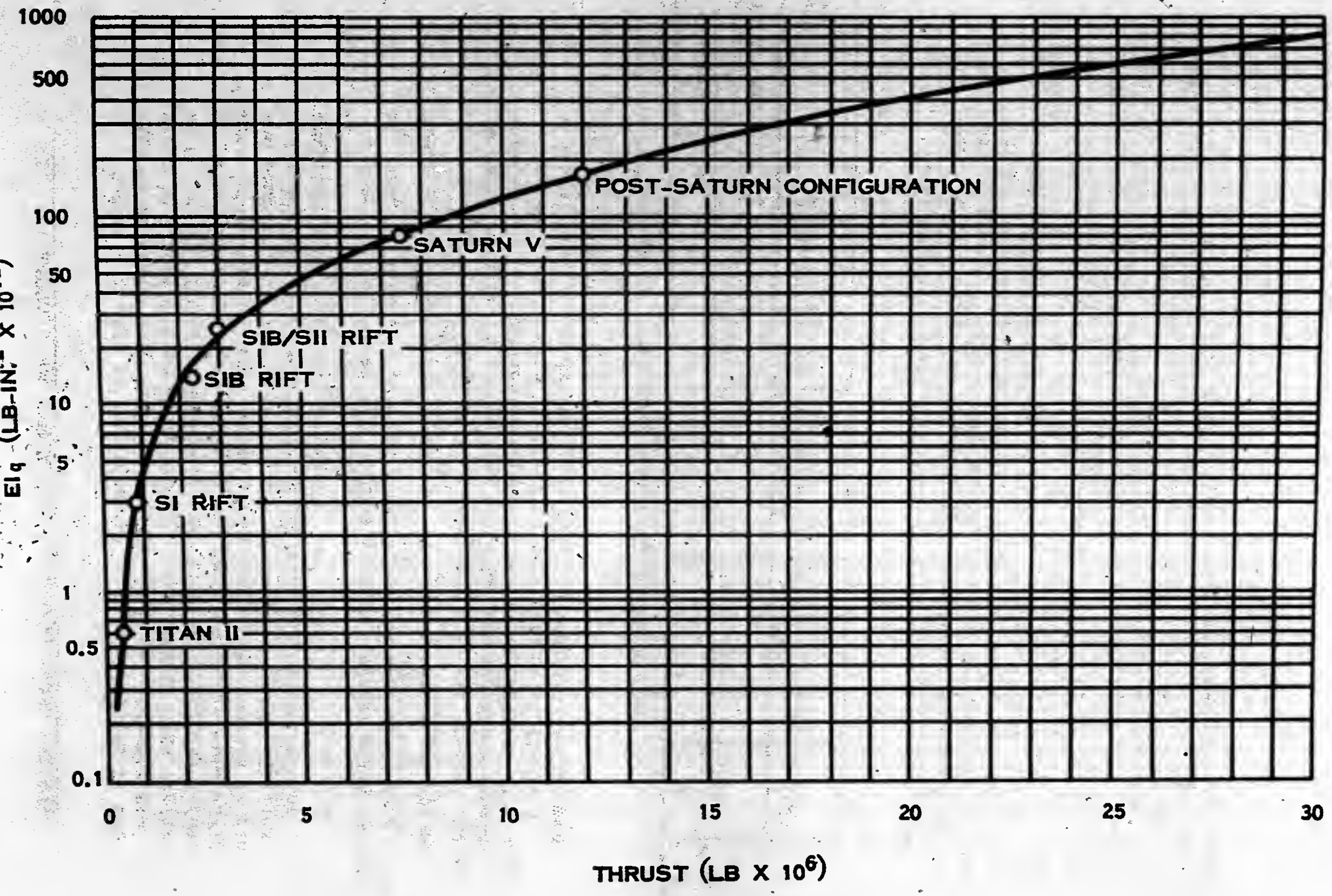

FIGURE 2 VEHICLE FLEXURAL STIFFNESS VS THRUST

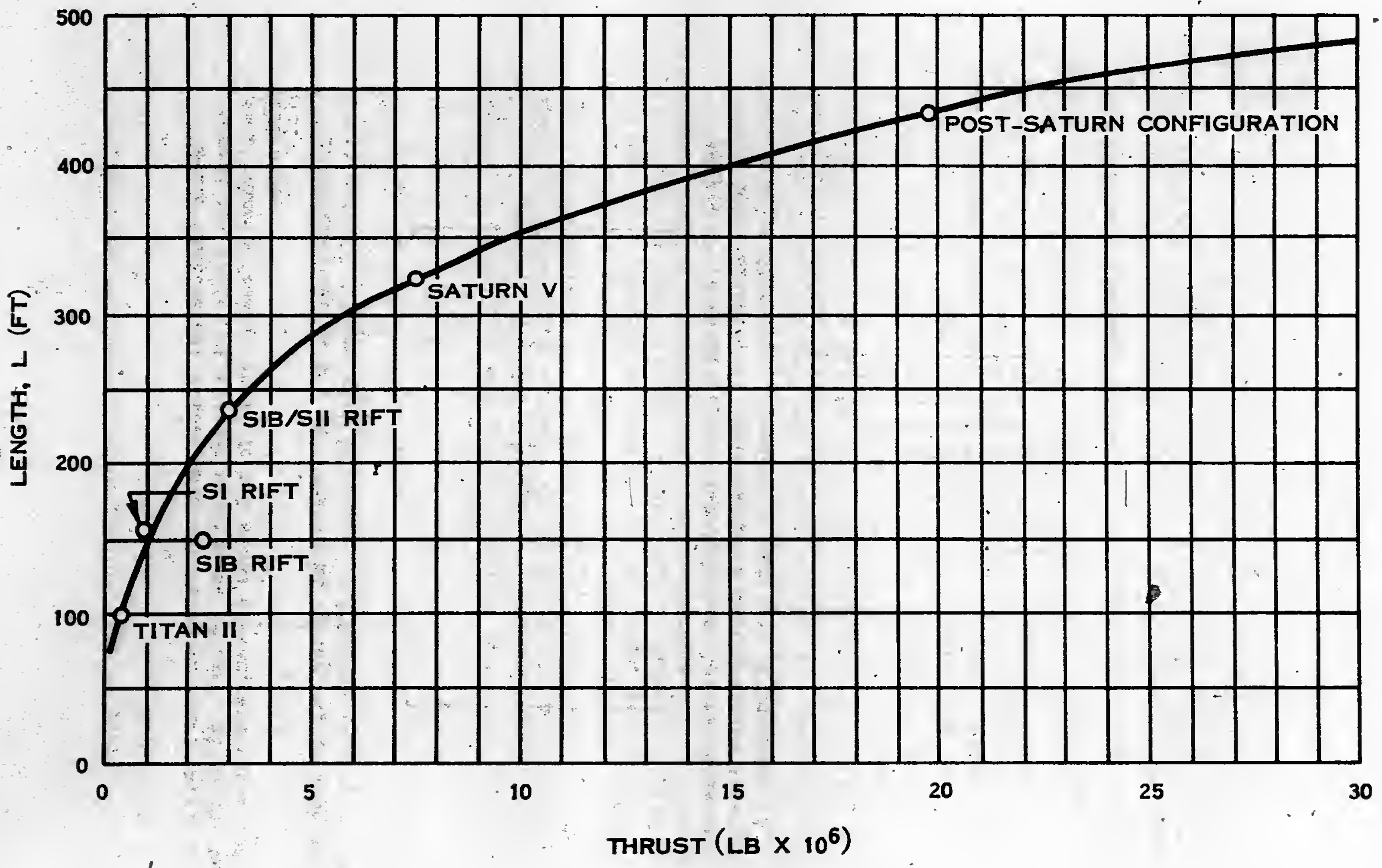

FIGURE 3 VEHICLE LENGTH VS THRUST

Notice that actual values need not be known since matrix A′ contains the ratios of average bending stiffnesses along the vehicle which of course apply to all sizes of vehicles under consideration.

In writing matrix B′, the rotational spring, R, is made infinitely stiff. Thus, with reference to Equation [4], $u_1 = 1$ and $u_2 = 0$.

$$
B' = \begin{bmatrix}
1 & -2 & 1 & & & & & & & \\
 & 1 & -2 & 1 & & & & & & \\
 & & 1 & -2 & 1 & & & & & \\
 & & & 1 & -2 & 1 & & & & \\
 & & & & 1 & -2 & 1 & & & \\
 & & & & & 1 & -2 & 1 & & \\
 & & & & & & 1 & -2 & 1 & \\
 & & & & & & & 1 & -2 & 1 \\
 & & & & & & & & 2 & -2 \\
 & & & & & & & & & \epsilon
\end{bmatrix}
$$

Matrix B′ contains first order difference patterns as well as the only variable, ϵ. This variable will be given several values after defining the remaining matrices of Equation [3].

$$
H = \begin{bmatrix}
1 & & & & & & & & & \\
2 & 1 & & & & & & & & \\
3 & 2 & 1 & & & & & & & \\
4 & 3 & 2 & 1 & & & & & & \\
5 & 4 & 3 & 2 & 1 & & & & & \\
6 & 5 & 4 & 3 & 2 & 1 & & & & \\
7 & 6 & 5 & 4 & 3 & 2 & 1 & & & \\
8 & 7 & 6 & 5 & 4 & 3 & 2 & 1 & & \\
9 & 8 & 7 & 6 & 5 & 4 & 3 & 2 & 1 & \\
1 & 1 & 1 & 1 & 1 & 1 & 1 & 1 & 1 & 1
\end{bmatrix}
$$

where H′ is a summing matrix.

M is a matrix of mass ratios. From Equation [2] and Table 1, it is seen that:

$$M = \begin{bmatrix} 1 &&&&&&&&& \\ & 1.33 &&&&&&&& \\ && 1.67 &&&&&&& \\ &&& 2.67 &&&&&& \\ &&&& 3.00 &&&&& \\ &&&&& 3.67 &&&& \\ &&&&&& 4.67 &&& \\ &&&&&&& 5.00 && \\ &&&&&&&& 5.00 & \\ &&&&&&&&& 5.33 \end{bmatrix}$$

The variable, ϵ, is a boundary spring parameter defined in Equation [1]. Five different values are assigned this variable to plot a dimensionless curve from which the upper and lower limits of lateral stiffness can be chosen. Experimental solutions indicate that for a practical range of the eigenvalue, λ_1, the highest value of ϵ should be 308, while the lowest value should be 0.0308 with factors of ten in between. Hence, five cases are matrically formulated from Equation [3] for solution on a digital computer. Ten eigenvalues and modes are found for each case with the primary interest directed toward only the first eigenvalue, λ_1. The square roots of these values are plotted against the boundary spring parameters, ϵ, in Figure 4. The square root of the eigenvalue is plotted because the frequency varies with the square root of λ. From this curve one may select an upper and lower limit of ϵ as shown in Figure 4. As a check, the fundamental frequency, f_1, is plotted against the stand lateral spring, k_h, which is shown in Figure 5 for a particular example where T = 3,000,000 pounds. This curve is applicable only for a vehicle having a booster thrust of 3,000,000 pounds, whereas the curve of Figure 4 is dimensionless and applicable for any thrust. The values of 4 and 9 for ϵ in Figure 4 are used in determining the lateral stand-stiffness requirements. From the definition of ϵ under Equation [1], one can compute the value of k_h for various thrusts since the length (or h) and base bending stiffness, EI_q, have been established as a function of thrust in Figures 2 and 3 respectively. The results of these computations are shown in columns 13 and 14 of Table 2 and also in Figure 6.

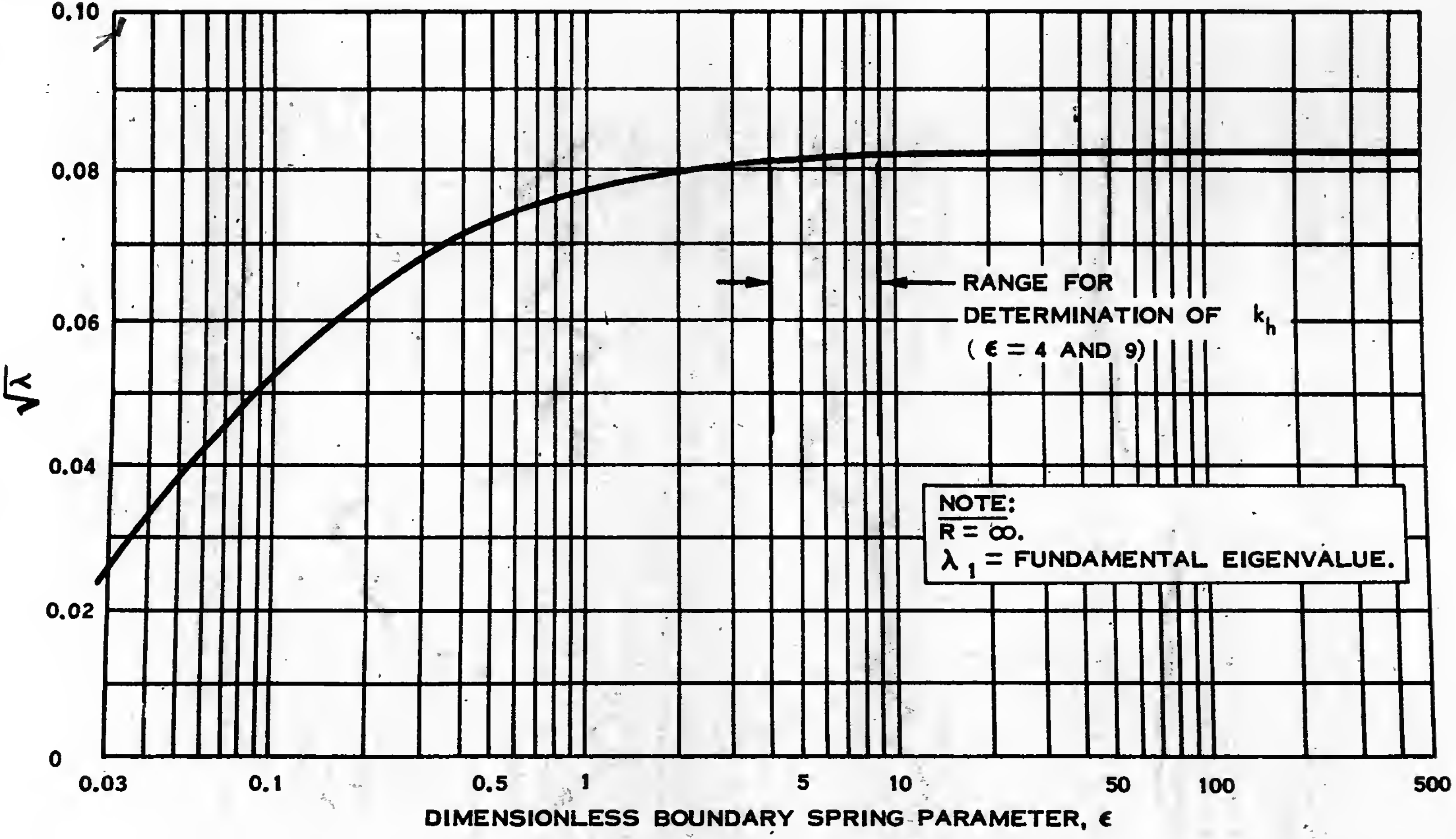

FIGURE 4 SELECTION OF LIMITS FOR k_h

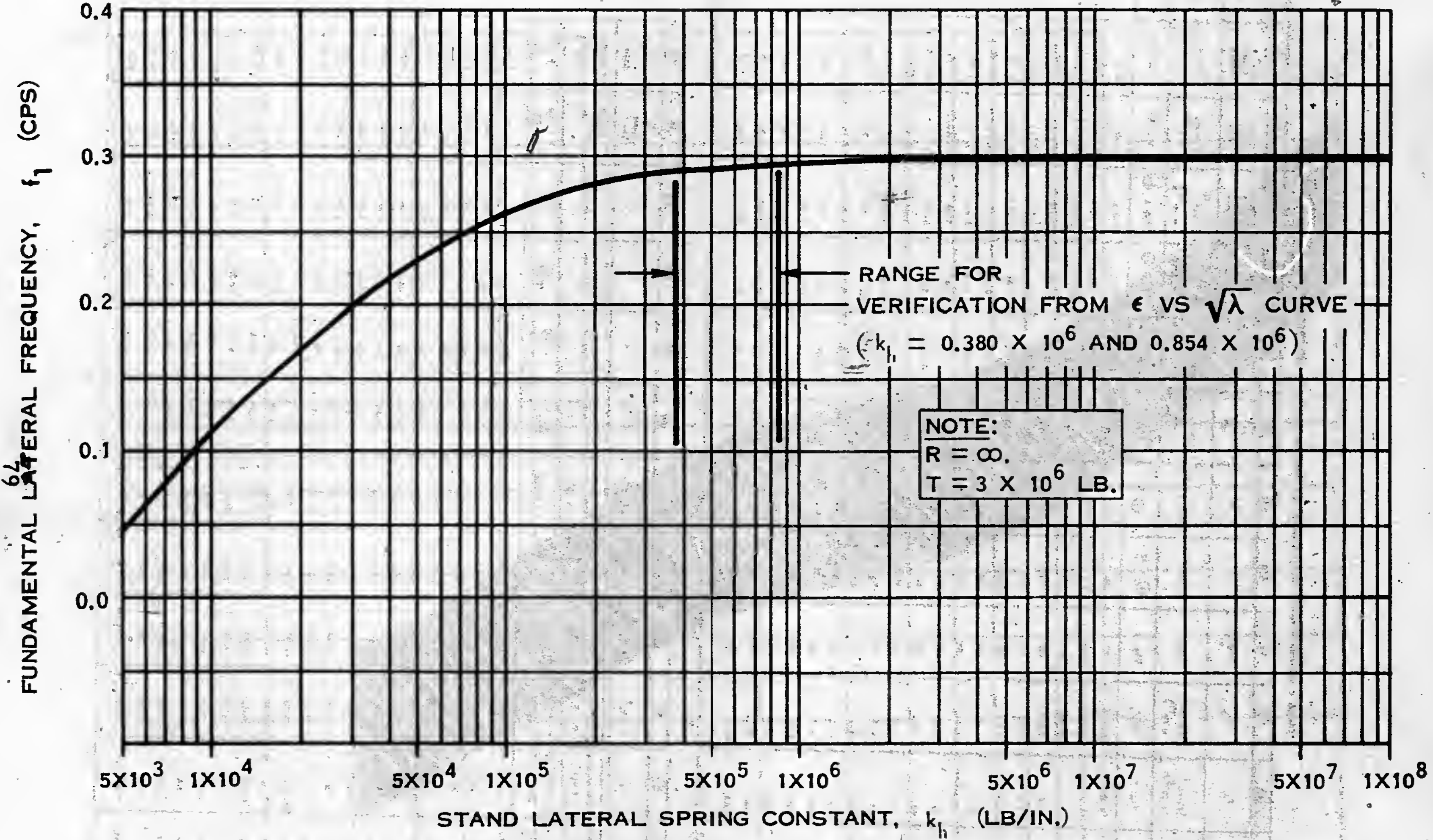

FIGURE 5 SELECTION OF LIMITS VERIFICATION

TABLE 6 COMPREHENSIVE DERIVATION OF ALL SPRING CONSTANTS AND FREQUENCIES

①	②	③	④	⑤	⑥	⑦	⑧	⑨	⑩	⑪	⑫	⑬	⑭	⑮
T	P	m_1	EI_0	EI_1	Length, L	h	h^3	$\frac{EI_1}{h^3}$	$\frac{EI_1}{h}$	K Upper Limit	K' Lower Limit	k_h Upper Limit	k_h Lower Limit	$\frac{EI_1}{m_1 h^3}$
(lb × 10⁶)	(lb × 10⁶)	$\left(\frac{\text{lb-sec}^2}{\text{in.}}\right)$	(lb-in.² × 10¹²)	(lb-in.² × 10¹²)	(ft)	(in.)	(in.³ × 10⁶)	(lb/in. × 10⁶)	(in.-lb × 10¹¹)	$\left(\frac{\text{in.-lb}}{\text{rad}} \times 10^{11}\right)$	$\left(\frac{\text{in.-lb}}{\text{rad}} \times 10^{11}\right)$	(lb/in. × 10⁶)	(lb/in. × 10⁶)	(1/sec²)
	①/1.3	$\frac{0.03 \times ②}{386}$	Fig. 2	④/8	Fig. 3	$\frac{⑥ \times 12}{9}$	⑦³	⑤/⑧	⑤/⑦	3.72 × ⑩	1.636 × ⑩	9 × ⑨	4 × ⑨	⑨/③
0.3	0.231	17.95	0.30	0.0375	75.5	100.7	1.02	0.0310	0.0281	0.103	0.046	0.279	0.124	1727.
0.6	0.463	35.9	1.3	0.1625	115.	153.3	3.56	0.0451	0.0848	0.315	0.139	0.406	0.180	1256.
0.9	0.692	53.8	3.1	0.388	141.	188.0	6.64	0.0584	0.1649	0.613	0.270	0.526	0.234	1085.
1.2	0.923	71.7	5.4	0.675	160.	213.	9.66	0.0699	0.234	0.863	0.416	0.629	0.280	975.
1.5	1.154	89.7	8.0	1.00	178	237	13.3	0.0753	0.339	1.26	0.555	0.677	0.301	839.
2.0	1.538	119.5	13.5	1.69	203	271	19.9	0.0849	0.498	1.85	0.815	0.764	0.340	710.
3.0	2.31	179.3	24.	3.0	237	316	31.6	0.0949	0.759	2.82	1.24	0.854	0.380	529.
4.0	3.08	239.4	35.	4.4	265	353	44.0	0.1000	0.992	3.69	1.62	0.900	0.400	418.
5.0	3.85	299.2	47.	5.9	285	380	54.9	0.1075	1.237	4.60	2.02	0.968	0.430	359.
6.0	4.61	359.1	58.	7.3	303	404	65.9	0.1130	1.485	5.52	2.43	1.02	0.452	317.
8.0	6.15	478.0	91.	11.4	330	440	85.2	0.1330	2.07	7.70	3.39	1.20	0.532	280.
10.0	7.69	597.7	129.	16.1	355	473	105.8	0.1522	2.73	10.16	4.47	1.37	0.609	255
12.0	9.23	717.4	170.	21.3	373	497	123.0	0.1736	3.42	12.72	5.60	1.56	0.694	242
14.0	10.77	837.0	218.	27.3	390	520	140.6	0.1935	4.19	15.58	6.85	1.74	0.774	233
16.0	12.31	956.7	270.	33.8	406	541	158.5	0.214	4.99	18.56	8.16	1.93	0.856	226
18.0	13.85	1076.	330.	41.2	420	560	175.6	0.235	5.89	21.9	9.64	2.12	0.940	219
20.0	15.38	1195.	400.	50.0	433	577	192.1	0.260	6.33	23.8	11.36	2.34	1.04	217.
25.0	19.23	1495.	580.	72.5	462	616	234.	0.320	9.73	36.2	15.90	2.88	1.28	213
30.0	23.08	1794.	800.	100.0	480	640	262.	0.382	12.30	46.4	20.5	3.44	1.53	212

⑯	⑰	⑱	⑲	⑳	㉑	㉒	㉓	㉔	㉕	㉖	㉗	㉘	㉙
ω_1^2	ω_1	f_h	r_1	r_1^2	k_2	k_v Upper Limit	k_v Lower Limit	k_2/m_1	ω_1^2	ω	f_v	K_T Upper Limit	K_T Lower Limit
(rad/sec²)	(rad/sec)	(cps)	(in.)	(in.² × 10⁴)	(lb/in. × 10⁶)	(lb/in. × 10⁶)	(lb/in. × 10⁶)	(1/sec² × 10³)	(rad/sec²)	(rad/sec)	(cps)	$\left(\frac{\text{in.-lb}}{\text{rad}} \times 10^{10}\right)$	$\left(\frac{\text{in.-lb}}{\text{rad}} \times 10^{10}\right)$
λ × ⑮ λ = 6.5 × 10⁻³	√⑯	⑰/6.283	Fig. 11	⑲²	$\frac{⑥}{⑦ ⑳}$	5 × ㉑	2 × ㉑	$\frac{㉑ \times 3}{③ \times 16}$	λ × ㉔ λ = 0.037	√㉕	$\frac{㉖}{2\pi}$	3 × ⑬ × ⑳	2 × ⑭ × ⑳
11.23	3.35	0.533	50	0.250	1.113	5.62	2.15	13.75	670.	25.3	4.03	0.140	0.062
8.16	2.86	0.455	72	0.518	1.637	8.18	3.27	8.55	487.	22.1	3.52	0.620	0.180
7.05	2.65	0.422	86	0.740	2.13	11.1	4.44	7.77	443.	21.0	3.35	0.780	0.346
6.34	2.52	0.401	95	0.902	2.61	14.0	5.62	7.35	413.	20.3	3.24	1.13	0.503
5.45	2.33	0.371	103	1.061	3.18	15.9	6.36	6.65	379	19.5	3.11	1.64	0.640
4.62	2.15	0.342	113	1.277	3.90	19.5	7.80	6.12	349	18.6	2.96	1.95	0.867
3.44	1.855	0.295	130	1.690	4.49	22.4	8.96	4.69	267	16.3	2.60	2.80	1.28
2.72	1.649	0.262	146	2.13	4.63	23.1	9.30	3.64	207	14.4	2.30	3.83	1.70
2.33	1.526	0.243	161	2.59	4.78	23.9	9.56	3.00	171.0	13.1	2.09	5.01	2.23
2.06	1.435	0.228	175	3.06	4.85	24.2	9.70	2.32	144.5	12.0	1.91	6.75	2.78
1.820	1.349	0.215	205	4.20	4.92	24.6	9.84	1.83	110.0	10.5	1.67	10.3	4.50
1.658	1.288	0.205	232	5.38	5.07	25.4	10.1	1.59	90.6	9.52	1.52	14.7	6.53
1.566	1.251	0.199	250	6.25	5.16	25.7	10.3	1.36	76.4	8.74	1.39	20.6	8.20
1.503	1.226	0.195	282	7.95	5.17	26.4	10.5	1.20	67.3	8.20	1.31	27.9	12.3
1.456	1.207	0.192	305	9.30	5.27	26.8	10.7	1.09	59.6	7.73	1.23	35.5	15.9
1.424	1.194	0.190	320	10.30	5.48	27.4	11.0	0.955	54.6	7.38	1.18	45.6	20.3
1.423	1.193	0.190	350	12.25	5.65	30.3	11.3	0.888	50.6	7.11	1.13	57.4	25.3
1.383	1.177	0.187	400	16.00	6.00	30.4	12.4	0.763	43.3	6.60	1.05	92.2	41.0
1.380	1.174	0.186	450	20.25	6.16	30.8	12.3	0.662	36.6	6.05	0.96	130.	67.0

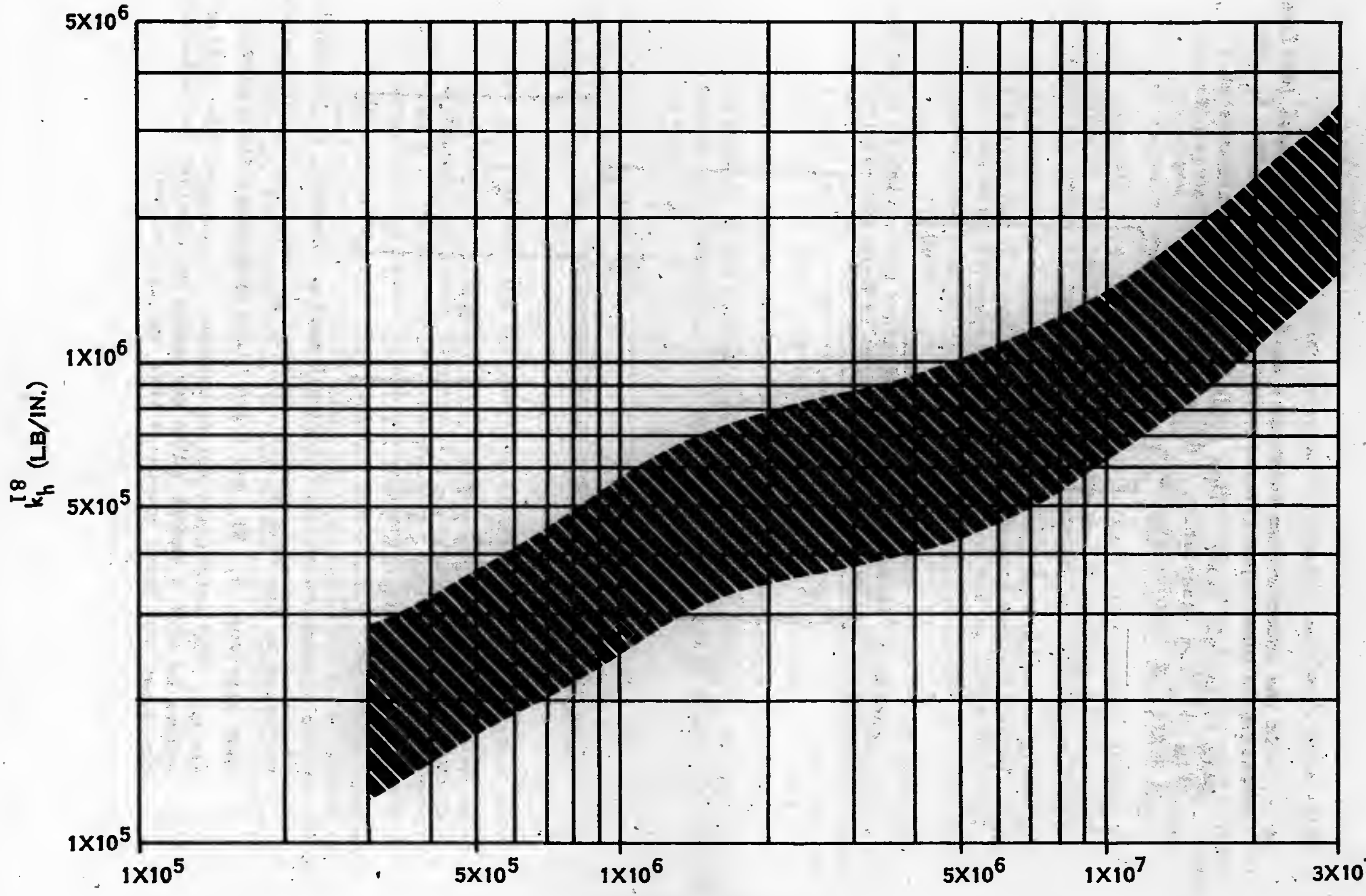

FIGURE 6 STAND LATERAL SPRING CONSTANT VS THRUST

In the solution for rotational stiffness, R, the same idealized vehicle is used. For this solution, the lateral spring, k_h, is made infinitely stiff. This condition restricts the mass at q (Fig. 1) from translation. Hence, this restriction results in a 9 degree of freedom system.

$$A = \begin{bmatrix} 1.33 &&&&&&&& \\ & 2.00 &&&&&&& \\ && 2.33 &&&&&& \\ &&& 4.00 &&&&& \\ &&&& 5.33 &&&& \\ &&&&& 6.67 &&& \\ &&&&&& 7.33 && \\ &&&&&&& 8.00 & \\ &&&&&&&& 8.00 \end{bmatrix}$$

Matrix B′ is somewhat different since the variable, ϵ, is replaced by a new variable, u_1, defined in Equation [4].

$$B' = \begin{bmatrix} 1 & -2 & 1 &&&&&& \\ & 1 & -2 & 1 &&&&& \\ && 1 & -2 & 1 &&&& \\ &&& 1 & -2 & 1 &&& \\ &&&& 1 & -2 & 1 && \\ &&&&& 1 & -2 & 1 & \\ &&&&&& 1 & -2 & 1 \\ &&&&&&& 1 & -2 \\ &&&&&&&& (1+u_1) \end{bmatrix}$$

$$H = \begin{bmatrix} 1 &&&&&&&& \\ 2 & 1 &&&&&&& \\ 3 & 2 & 1 &&&&&& \\ 4 & 3 & 2 & 1 &&&&& \\ 5 & 4 & 3 & 2 & 1 &&&& \\ 6 & 5 & 4 & 3 & 2 & 1 &&& \\ 7 & 6 & 5 & 4 & 3 & 2 & 1 && \\ 8 & 7 & 6 & 5 & 4 & 3 & 2 & 1 & \\ 9 & 8 & 7 & 6 & 5 & 4 & 3 & 2 & 1 \end{bmatrix} \qquad M = \begin{bmatrix} 1 &&&&&&&& \\ & 1.33 &&&&&&& \\ && 1.67 &&&&&& \\ &&& 2.67 &&&&& \\ &&&& 3.0 &&&& \\ &&&&& 3.67 &&& \\ &&&&&& 4.67 && \\ &&&&&&& 5.0 & \\ &&&&&&&& 5.0 \end{bmatrix}$$

u_1 is a function of the rotational spring constant, R, the bending stiffness at the base of the vehicle, EI_q, and the spacing between mass stations, h. EI_q and h are fixed quantities for a given thrust. The rotational spring constant is assigned five different values so that a smooth curve may again be obtained for selecting the upper and lower bounds of this spring constant.

It is obvious from Equation [4] that u_1 may vary from -1.0 to +1.0. Five cases were matrically formulated for solution on a digital computer. As in the previous parameter study, the formulation and solution of five different cases is very rapid since only one element of the B' matrix requires different values. Again, ten eigenvalues and modes are found for each of the five cases. A dimensionless plot of u_1 vs $\sqrt{\lambda_1}$ (where λ_1 is the fundamental eigenvalue) is shown in Figure 7. Due to the gradual change in slope of the right side of this curve, it is very difficult to select definite limits for the parameter, u_1. Hence, a plot of fundamental frequency vs the rotational spring, R, is given in Figure 8 for T = 3,000,000 pounds. A more definite slope change is seen here from which the limits of R are selected and also established for the dimensionless plot. The values of 1.24×10^{11} and 2.28×10^{11} correspond with values for u_1 of -0.10 and 0.30, respectively. These latter parameter values may be used to determine the rotational spring stiffness requirements of launch stands for the entire range of vehicle thrusts considered. From Equation [4], for the upper and lower values of u_1, one may calculate R for different thrust values since EI_q and L (and hence h) are given in Figures 2 and 3 as a function of thrust. These R values vs thrust are computed in columns 11 and 12 of Table 2 and are plotted in Figure 9.

III. DETERMINATION OF VERTICAL SPRING CONSTANTS

In the solution for vertical stiffness, k_v, a slightly different approach is used in the development of the required vertical stiffness for launch stands. The idealized model is shown in Figure 10 which considers longitudinal motion only. The vertical spring in the launch stand is represented by k_v, while longitudinal vehicle springs exist between each mass. The matric formulation for this problem is based on a method given in Reference (2). Essentially, a matric formulation is developed for the free vibrations of a linearly coupled vibrating system of many springs and masses. For application to this problem, the equations of motion may be expressed as:

$$(C - \lambda D) X = 0 \quad . \quad . \quad . \quad . \quad . \quad . \quad . \quad [5]$$

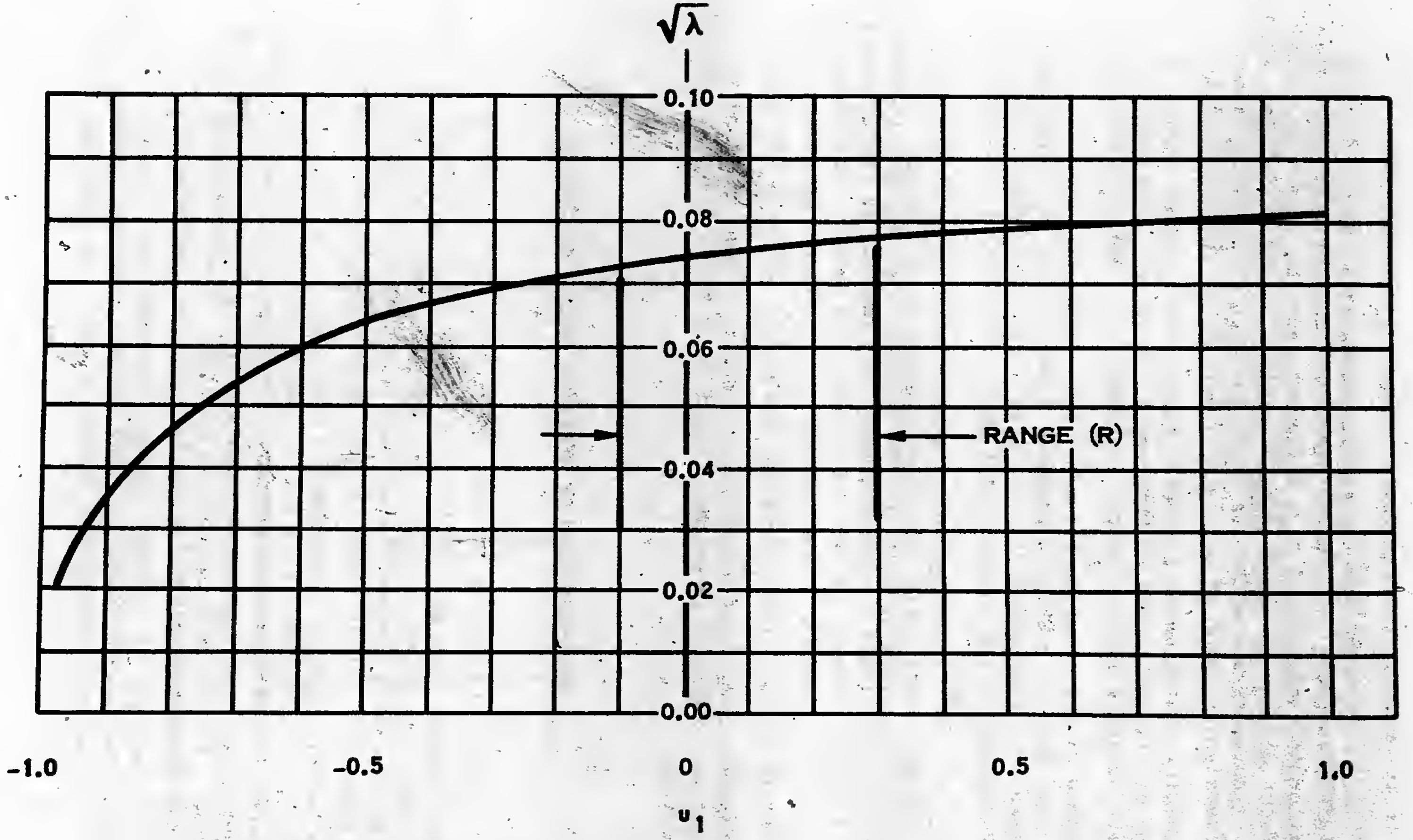

FIGURE 7 DIMENSIONLESS PLOT OF $\sqrt{\lambda}$ VS u_1 ($k_h = \infty$)

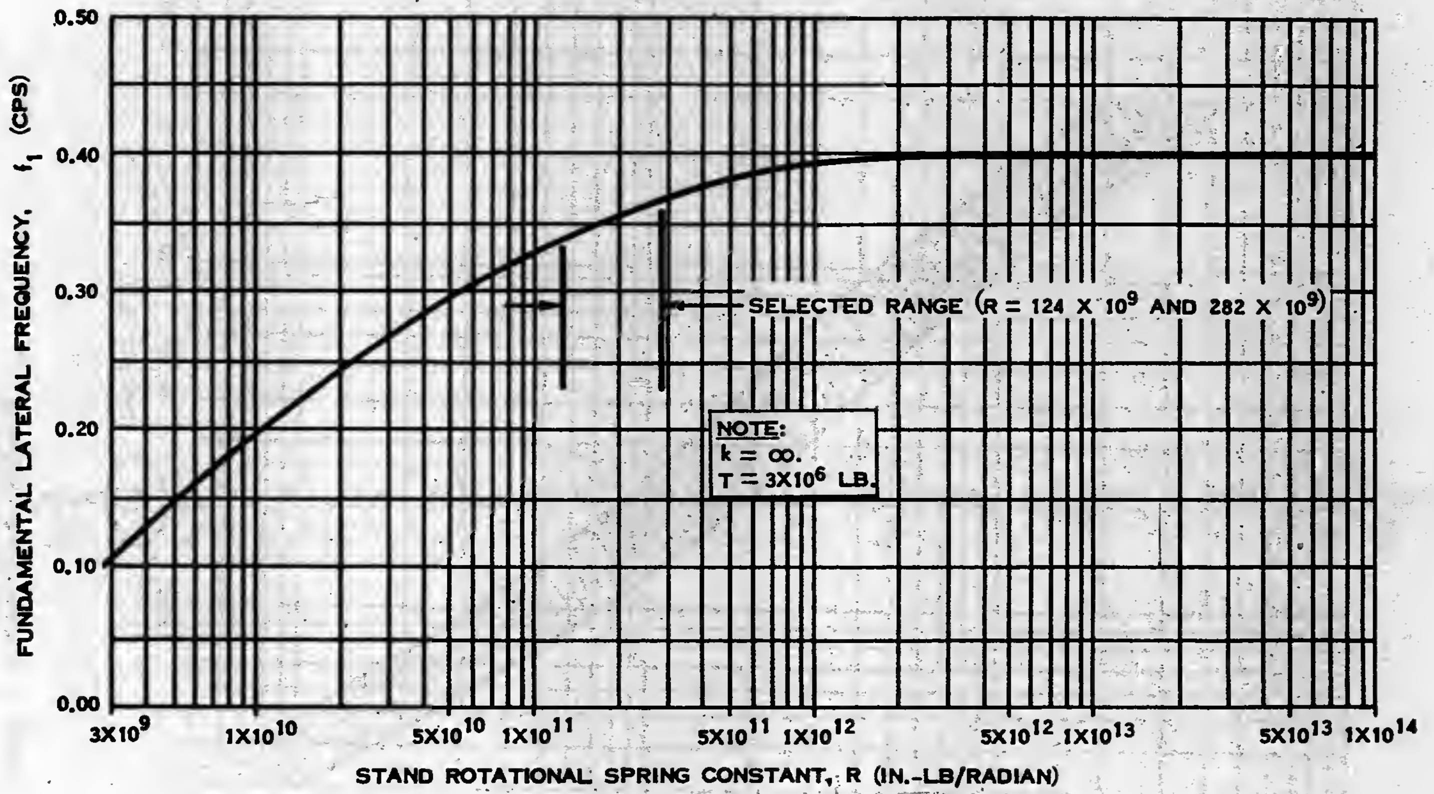

FIGURE 8 SELECTION OF LIMITS FOR R

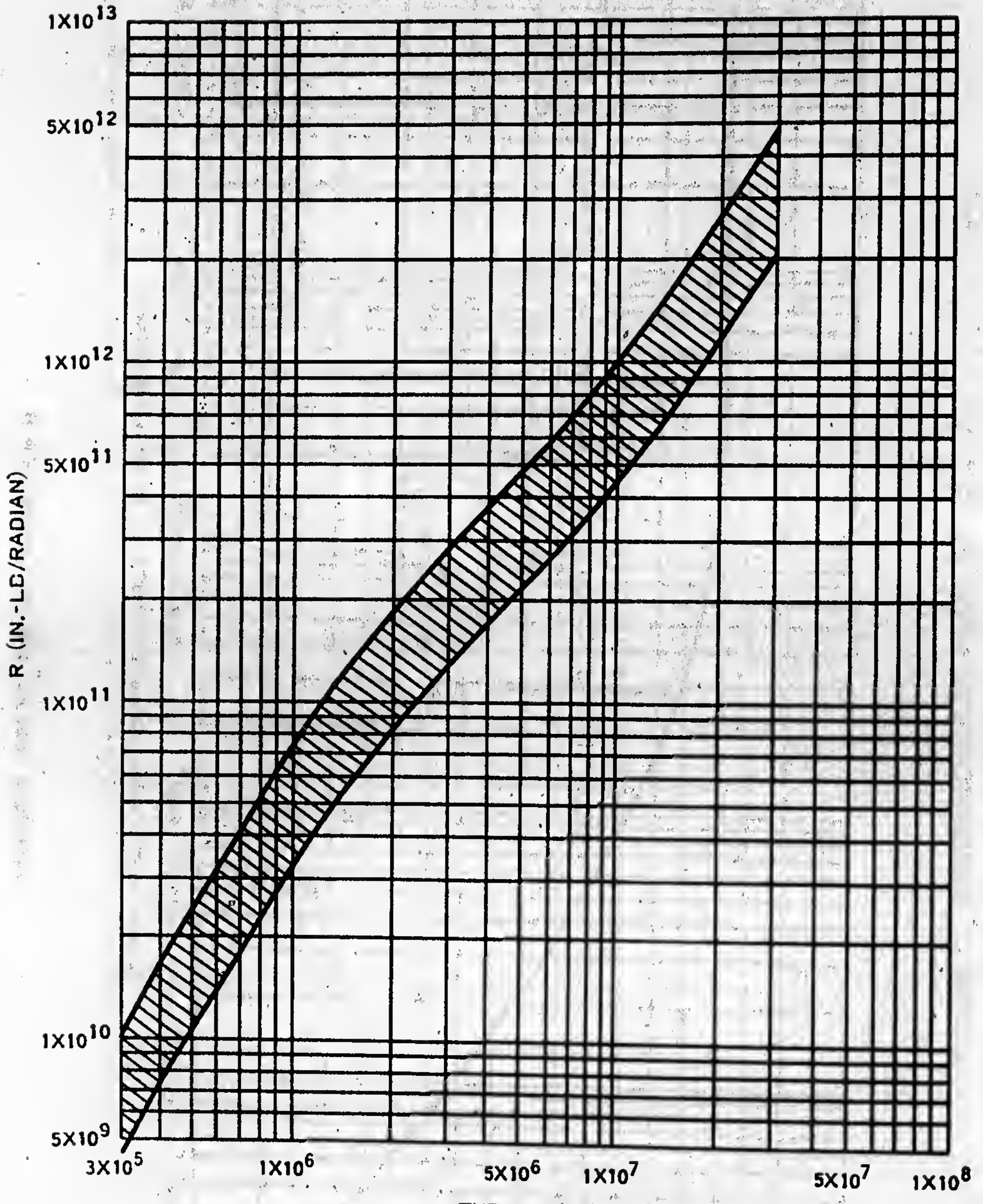

FIGURE 9 STAND ROTATIONAL SPRING CONSTANT VS THRUST

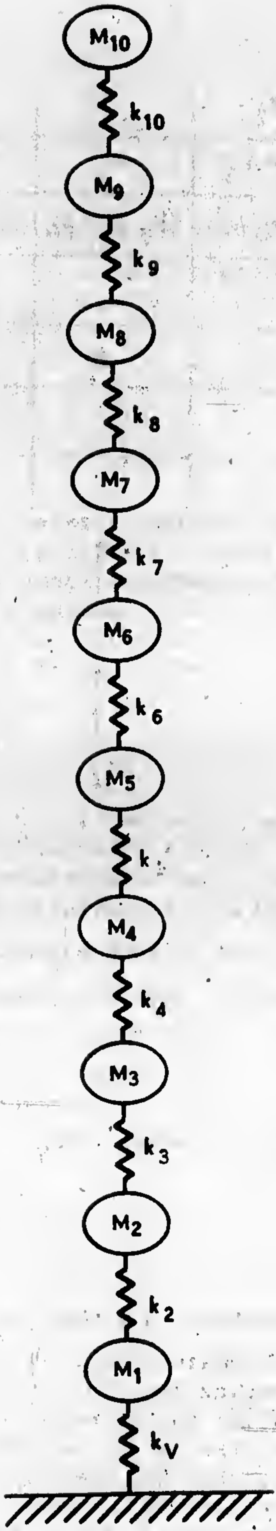

FIGURE 10 IDEALIZED MODEL FOR VERTICAL SPRING DETERMINATION

$$C = \begin{bmatrix} (k_v + k_2) & -k_2 & & & & & \\ -k_2 & (k_2 + k_3) & -k_3 & & & & \\ & -k_3 & (k_3 + k_4) & -k_4 & & & \\ & & \cdot & \cdot & \cdot & & \\ & & & \cdot & \cdot & \cdot\;\cdot & \\ & & & & -k_9 & (k_9 + k_{10}) & -k_{10} \\ & & & & & -k_{10} & k_{10} \end{bmatrix}$$

$$D = \begin{bmatrix} m_1 & & & & & \\ & m_2 & & & & \\ & & \cdot & & & \\ & & & \cdot & & \\ & & & & \cdot & \\ & & & & & m_{10} \end{bmatrix}, \qquad X = \begin{bmatrix} x_1 \\ x_2 \\ x_3 \\ \cdot \\ \cdot \\ \cdot \\ x_{10} \end{bmatrix},$$

where C is the spring matrix, D is the mass matrix, and X is the eigenvector. Since a parameter study is desired, both the C matrix and the D matrix can be expressed in nondimensional form. If each element in matrix C is divided by k_2 with the resulting matrix designated C' and if each element in matrix D is divided by m_1 with the resulting matrix designated D', then Equation [5] may be expressed as:

$$(C' - \lambda D')\, X = 0 \; . \; . \; . \; . \; . \; . \; . \; . \; . \; . \; . \quad [6]$$

The nondimensional eigenvalue, λ, is now expressed as:

$$\lambda = \frac{m_1}{k_2}\, \omega^2 \; . \; . \; . \; . \; . \; . \; . \; . \; . \; . \; . \; . \quad [7]$$

The spring constant, k_2, must be found as well as the relative values of the remaining springs. Longitudinal springs in a missile are a complex function of tank dome flexibility as well as axial deformation in the outer ring and stringers. As a general approximation, the spring constant is based on the deformation of a thin ring that has the same area as the skin stringer combination but is reduced by a factor to compensate for tank bottom deformation. For a cylinder of length, h, and having a relatively small thickness, the longitudinal spring constant may be expressed as:

$$k_i = \frac{2EI_i}{hr_i^2} \qquad i = 2 \;.....\; 10, \qquad [8]$$

where, EI_i is the average stiffness of the section and r_i is the radius of the particular section. To account approximately for tank dome flexibilities, Equation [8] may be reduced by a factor of 2. Hence,

$$k_i = \frac{EI_i}{hr_i^2} \qquad i = 2 \;.....\; 10. \qquad [9]$$

Based on actual base diameters of existing missiles and space vehicles, the radius at the base of vehicles is shown in Figure 11 as a function of the thrust range under consideration. Current design practice has shown that the ratios of vehicle longitudinal springs should be approximately like that shown in matrix C´. The value of λ in Equation [6] can be determined with k_v/k_2 as the only variable. The matrices are:

$$C' = \begin{bmatrix} \left(\frac{k_v}{k_2} + 1\right) & -1 & & & & & & & & \\ -1 & 2 & -1 & & & & & & & \\ & -1 & +2 & -1 & & & & & & \\ & & -1 & +2 & -1 & & & & & \\ & & & -1 & 1.75 & -0.75 & & & & \\ & & & & -0.75 & 1.50 & -0.75 & & & \\ & & & & & -0.75 & 1.50 & -0.75 & & \\ & & & & & & -0.75 & 1.25 & -0.50 & \\ & & & & & & & -0.50 & 1.00 & -0.50 \\ & & & & & & & & -0.50 & 0.50 \end{bmatrix}$$

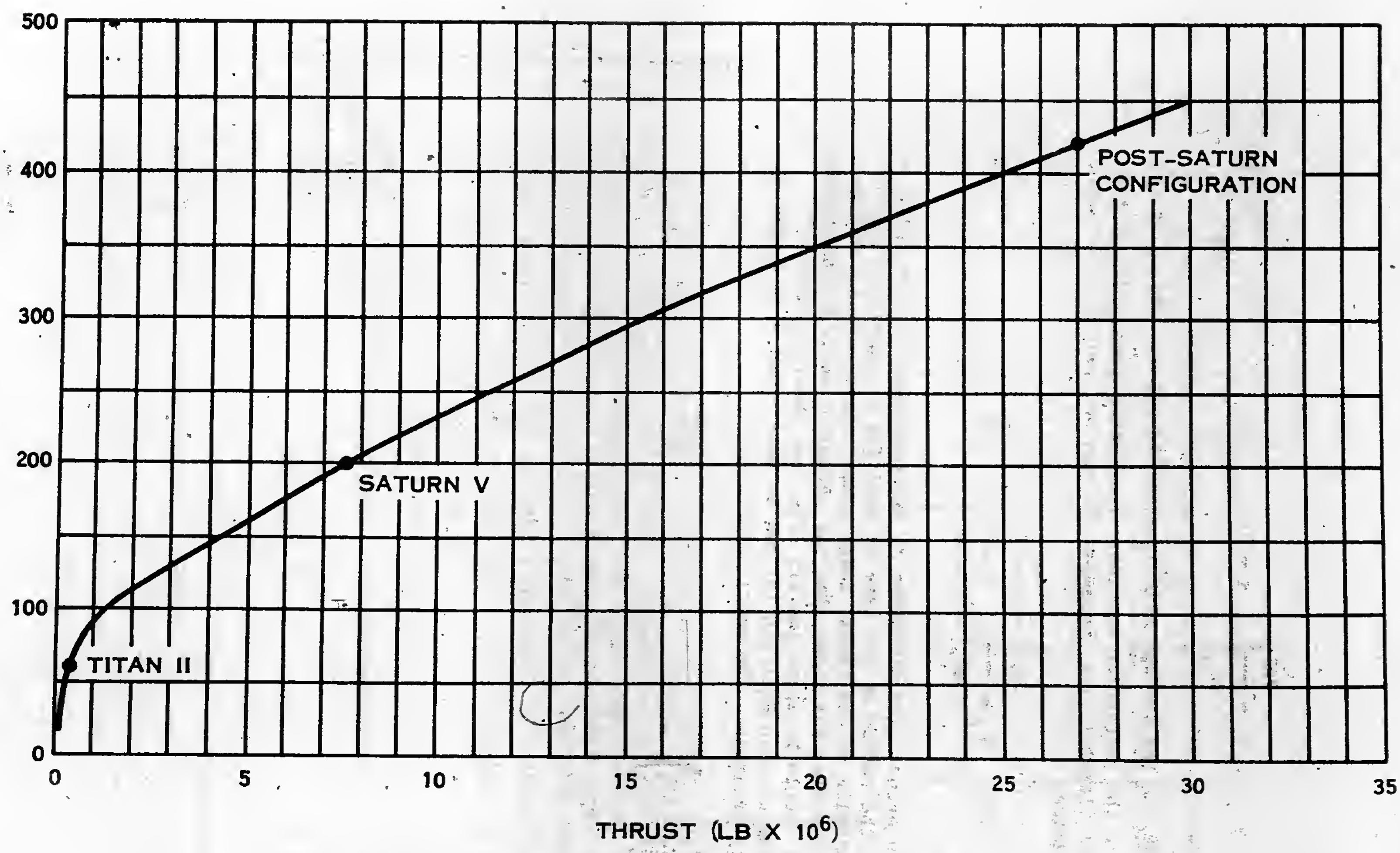

FIGURE 11 VEHICLE BASE RADIUS VS THRUST

$$D' = \begin{bmatrix} 1.0 &&&&&&&&& \\ & 0.9375 &&&&&&&& \\ && 0.0375 &&&&&&& \\ &&& 0.875 &&&&&& \\ &&&& 0.6875 &&&&& \\ &&&&& 0.5625 &&&& \\ &&&&&& 0.50 &&& \\ &&&&&&& 0.3125 && \\ &&&&&&&& 0.25 & \\ &&&&&&&&& 0.1875 \end{bmatrix}$$

where D' is based on the same mass ratios used in Table 1.

Values of the ratios of k_v / k_2 have been assumed from 0.12 through 100, and the corresponding values of λ have been determined from Equation [6] for a selected number of ratios of k_v / k_2. The results of the solutions and the range for determining k_v are shown in Figure 12. Values of k_2 can be determined from Equation [9] by reference to Figures 2, 3, and 11 for various thrust values. From Figure 12, the selected lower and upper limits of k_v / k_2 are 2 and 5. Therefore, since k_2 is known as a function of thrust, k_v may be established as a function of thrust for the lower and upper limits of 2 and 5. Calculations for k_v are shown in columns 22 and 23 of Table 2 and finally in Figure 13.

IV. DETERMINATION OF TORSIONAL SPRING CONSTANTS

It can be shown that the total torsional spring constant at the vehicle attachment points is:

$$R_T = 2k_h r^2 \qquad [10]$$

Using the above relationship and the results of Figures 11 and 6, one can develop Figure 14 which shows the torsional spring constant requirements vs thrust.

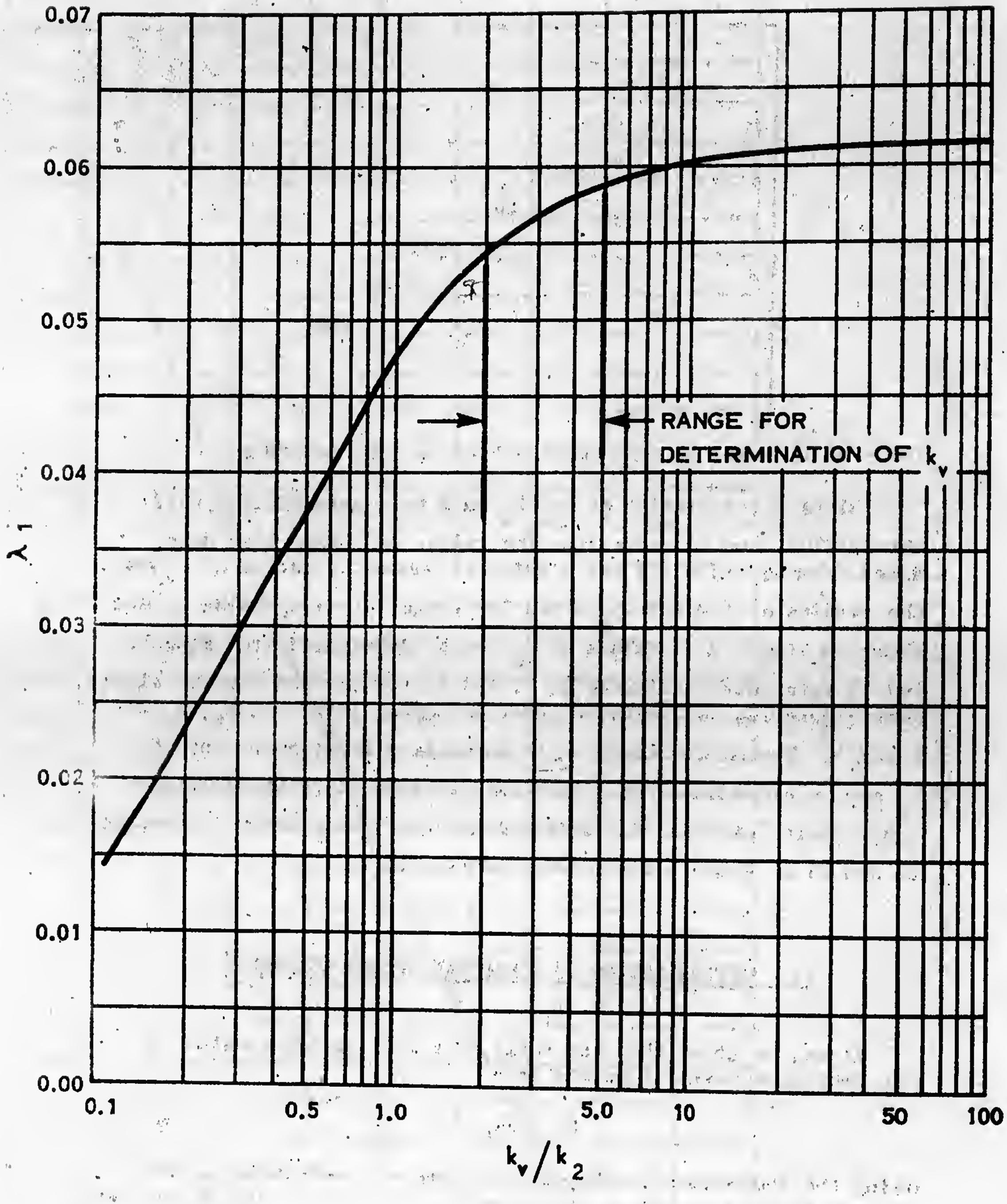

FIGURE 12 SELECTION OF LIMITS FOR k_v

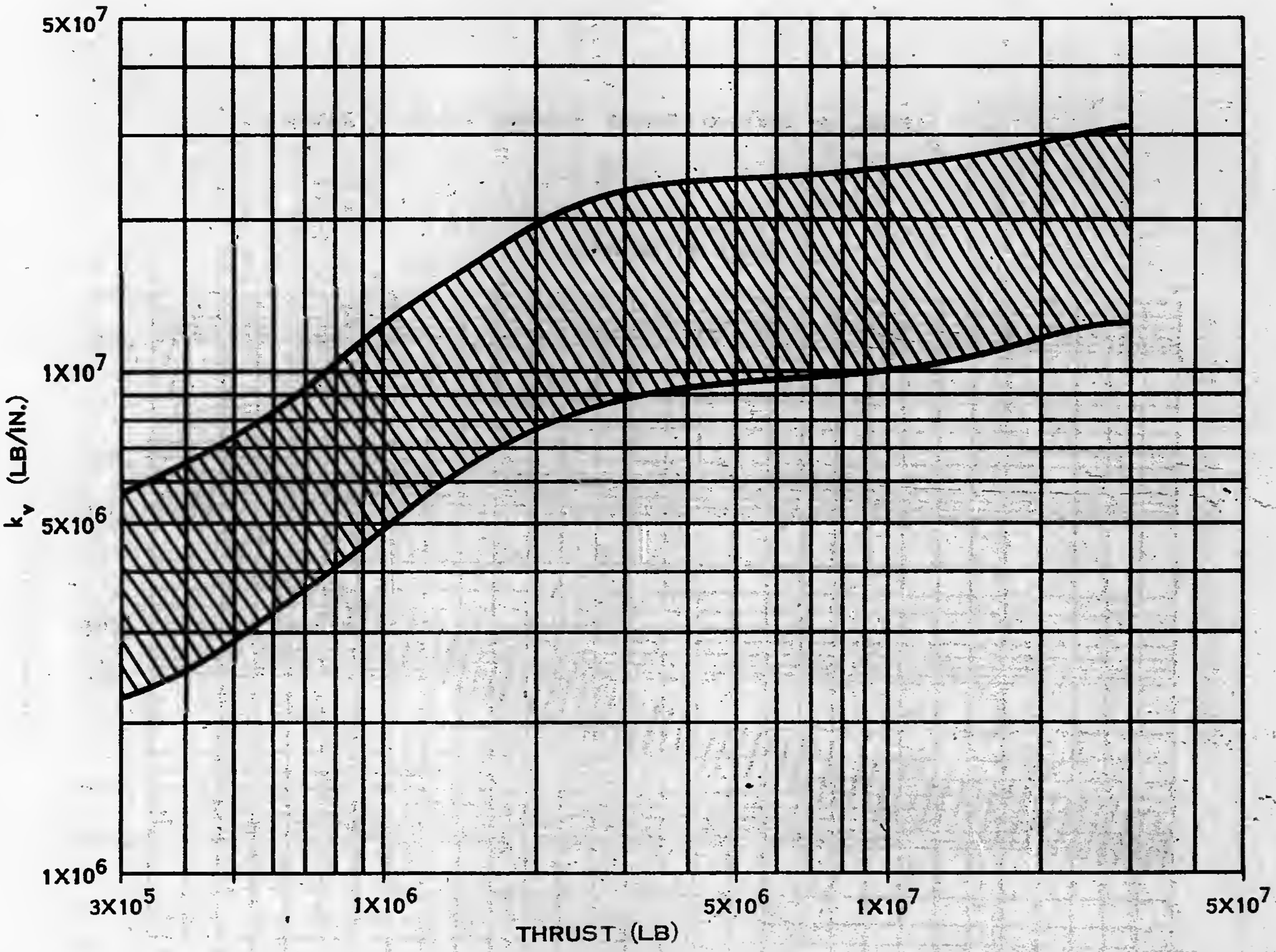

FIGURE 13 STAND VERTICAL SPRING CONSTANT VS THRUST

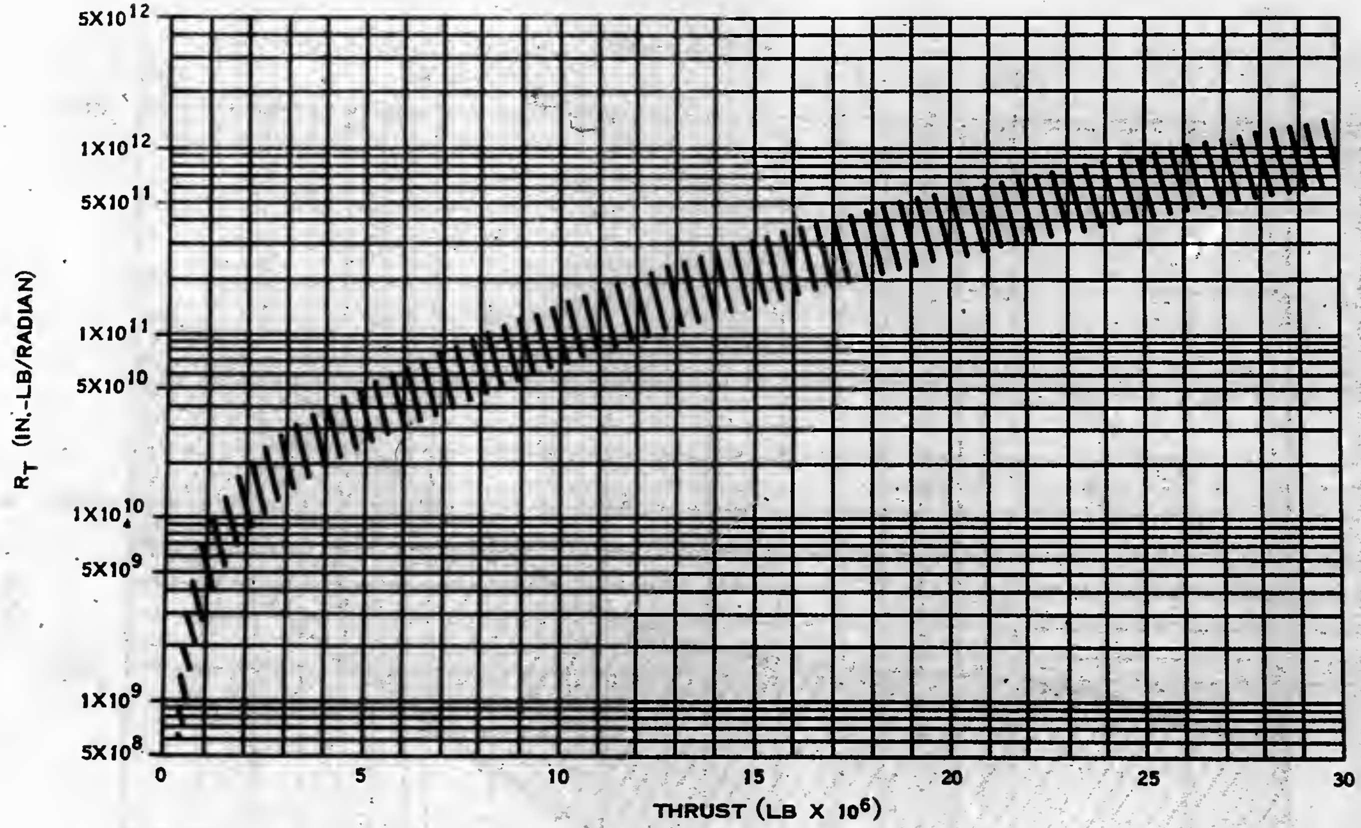

FIGURE 14 STAND TORSIONAL SPRING CONSTANT VS THRUST

V. FUNDAMENTAL FREQUENCY CONSIDERATIONS

Although the primary objective of this paper is to establish the overall total stiffness requirements that should exist at the vehicle attachment points for a large range of booster thrusts, the prediction of fundamental structural frequencies may also be of interest to the vehicle designer. From the expression for λ under Equation [1], and using 0.0065 as an approximate value for λ_1, obtained from Figure 4, one can easily find the lateral fundamental frequency of the vehicle-stand combination for any thrust. The calculations are made in Columns 15 thru 18 in Table 2, and the results are shown in Figure 15. Similarly from Equation [7] and Figure 12, one can solve for the vertical fundamental frequency of the system shown in Figure 10 for any thrust. These calculations are made in columns 24 through 27 of Table 2 while the results are shown in Figure 16.

VI. CONCLUSIONS

Once the approximate size of a proposed launch vehicle has been established, the launch or test stand design engineer may establish the thrust-mount stiffness requirements at the vehicle attachment points simply by referring to Figures 6, 9, 13, and 14. Therefore, design of the launch stand can proceed long before the final flight vehicle mass and stiffness data are available. Experience on the Titan and Saturn programs has shown that the stiffness criteria requirements presented in this paper provide the flight vehicle with economically and technically sound boundary conditions. For example, these stiffness criteria were used early in 1963 for establishing the Saturn V launcher stiffness requirements prior to design of the launcher-umbilical tower as given in Reference (3). Detailed analysis has shown that the dynamic loads imposed on the flight vehicle from wind, engine start malfunctions, and engine gimbaling will not be unreasonably excessive when the launch stand stiffness requirements recommended herein are met.

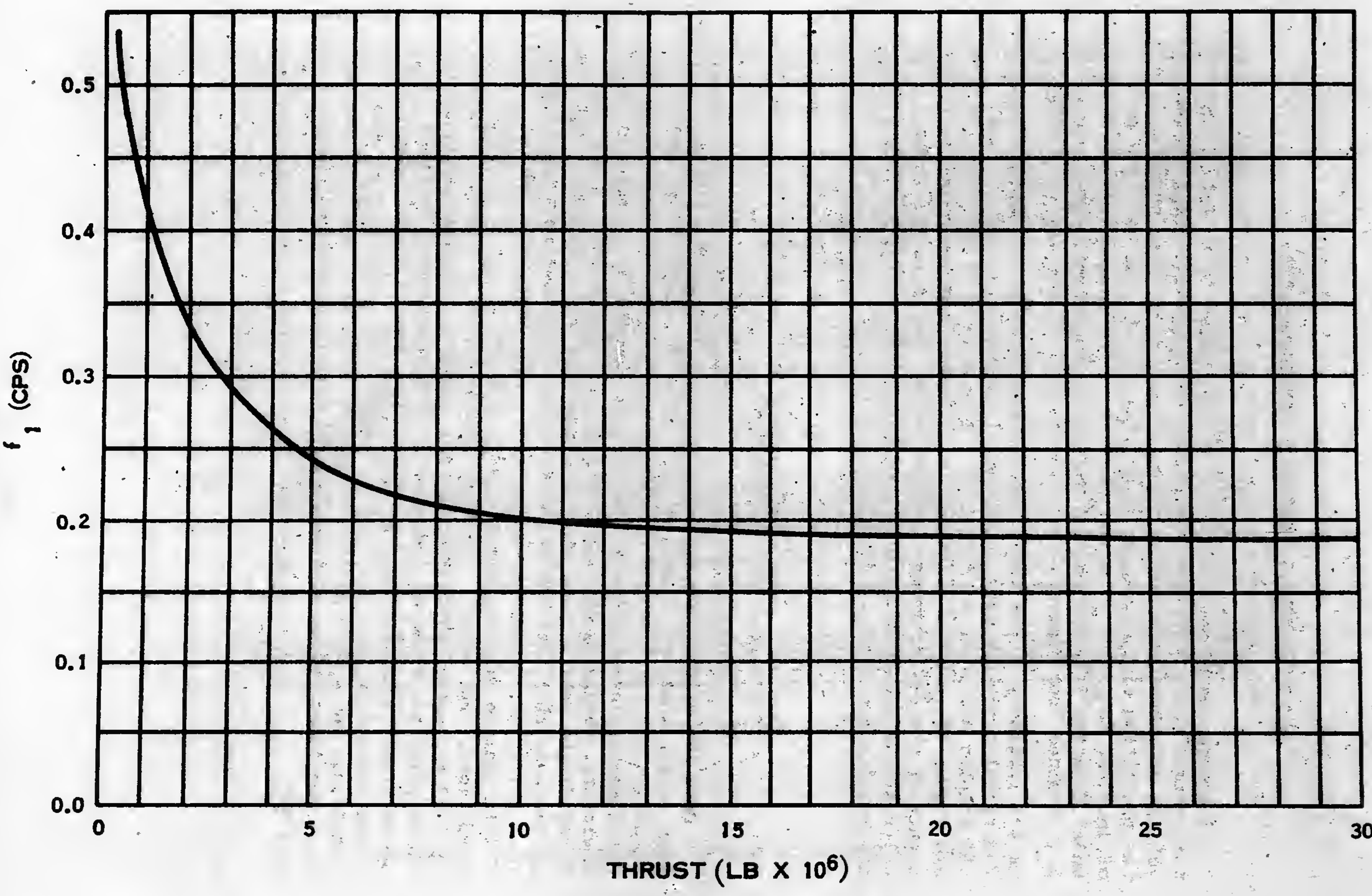

FIGURE 15 FUNDAMENTAL LATERAL FREQUENCY VS THRUST

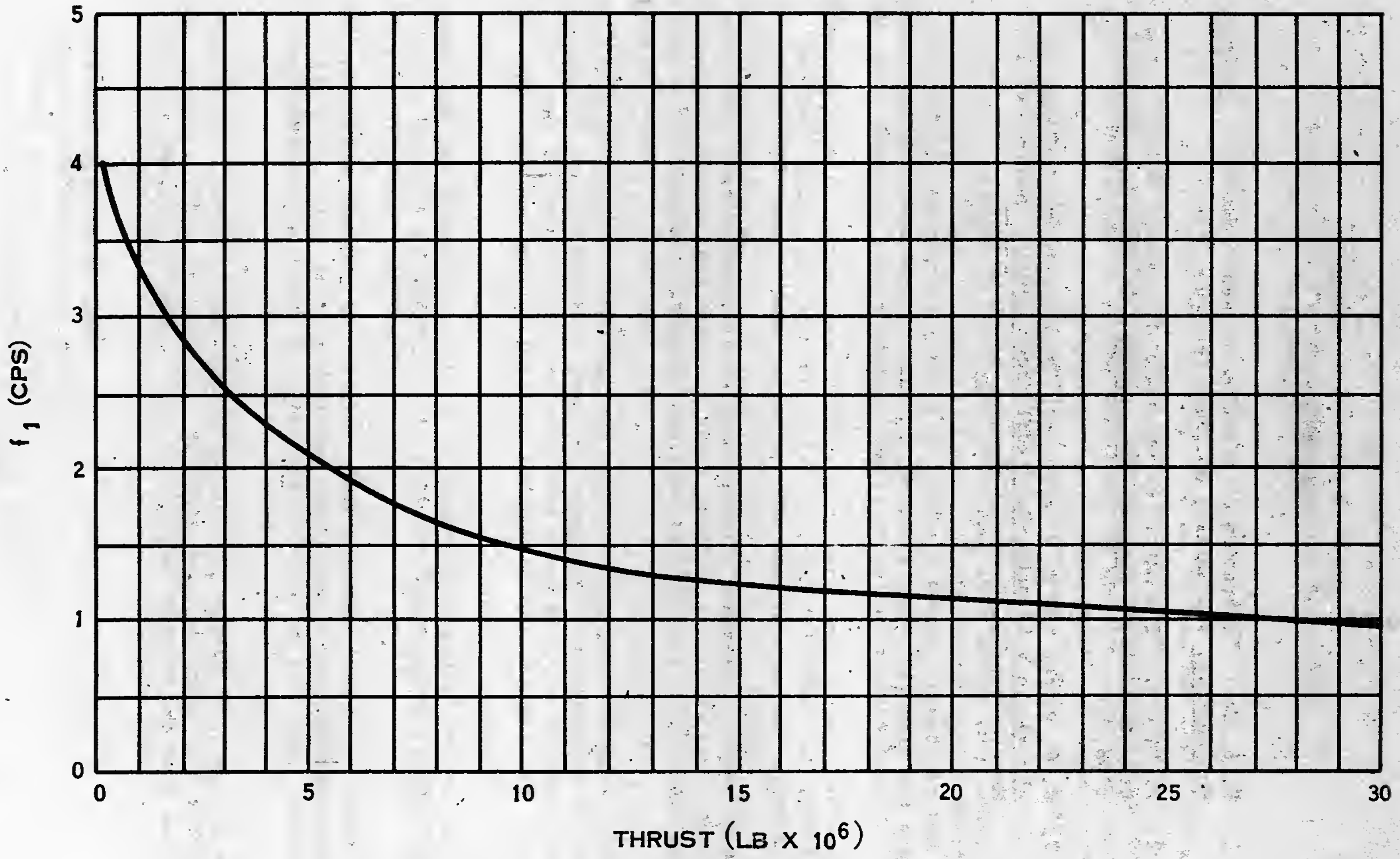

FIGURE 16 FUNDAMENTAL VERTICAL FREQUENCY VS THRUST

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GROUND SUPPORT FOR MAN IN SPACE

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Washington, D. C.

Since man first conceived the idea of rockets, he has been concerned with facilities to launch them, and engineers have had to design and build them. Initially, they were relatively simple. However, the rapid intensification of manned space flight efforts in recent years has led to even larger and more sophisticated launch vehicles, and corresponding developments in launch concepts. The result has been a vast ground support effort for man in space -- itself a major challenge of the national space effort.

As those of us engaged in aerospace activities know better than most, this Nation has made giant strides in the aerospace field since the end of World War II, when the Army started testing variations of captured German rockets at the White Sands Missile Range in New Mexico. During the years that followed, the simple rocket has been developed into a complex family of ballistic missiles.

The most advanced are the great ICBM's which can carry nuclear warheads to targets a continent away. The Army Engineers have built hundreds of launchers for these, of several different types, as the weapons and their ground support evolved.

It was highly specialized work -- an engineer's dream which often became a nightmare -- a tremendous team effort without parallel in the Nation's history of construction which received fitting recognition in 1962 as the outstanding Civil Engineering achievement of the year.

Today, with hundreds of ICBM's on the alert, we are over the hump in the urgent, nationwide program which so greatly strengthened the defenses of this country. The same technology that achieved this vital military success also ushered in the Space Age and made it possible for man to set new goals far beyond our earth.

A space age revolution is sweeping our country, propelled by the impetus of The National Aeronautics and Space Administration's Manned Space Flight Program as it builds up to its great climax.

A few years from now NASA will launch three astronauts in lunar orbit 100 miles above the moon. Once in this "Parking Orbit" two of the spacemen will leave their Apollo mother ship and head for the lunar surface in a 12 ton, bug-shaped shuttle.

They will land, perform their exploring mission, rejoin their Apollo vehicle and return to earth. When this has been accomplished, they will have fulfilled one of mankind's most ambitious dreams. Yet, the logical sequels are even more far-reaching:

A manned base on the moon,
Manned flights to other planets,
Unmanned probes of other parts of our universe,
Astronomical observatories orbiting in space,

A vital element of this Space Program is the development of a vast array of rocket and spacecraft fabrication and assembly plants, as well as ground-testing and launching facilities, in a year-round warm climate.

Here, along a 1500 mile crescent starting on a lake front in Eastern Texas down below Houston and sweeping through the South to a swampland on Florida's Banana River, are the major NASA Centers which are engaged in the Manned Space Flight effort. Their work has been called the beginning of a new and basic industry, sired by the missile age for the 21st Century. But it's also an industry based on solid construction, for without adequate construction support this program could never be undertaken.

The Army Engineers are playing a major role in this effort. Our part includes a number of different activities. However, at this stage of this rapidly moving space program our most important contribution is a massive construction buildup. (Figure 1)

Today our space projects include complex facilities at a number of locations for the National Aeronautics and Space Administration and the Air Force's Space Systems Division. Examples are:

1. Rocket engine test stands in California, and
2. Test facilities for the Lunar Excursion Vehicle at WSMR, New Mexico.

However, the bulk of our space work is at the four Centers on the Gulf Crescent --

1. The Manned Spacecraft Center at Houston which develops the spacecraft.
2. The Marshall Space Flight Center at Huntsville, Alabama which develops the rocket engines.
3. The quarter billion dollar facility in Mississippi for testing the space boosters.
4. The billion dollar spaceport at Cape Kennedy.

In part, the assignment of this critical work to the Corps derives from our vast experience in large and critical construction. But it also reflects the dependence of this Space Program on the navigable waterway system, which the Corps has developed over the years. (Figure 1). These four key Centers all adjoin our waterways, since the giant boosters of the space program are too large to travel except by water.

The Air Force's Atlas and Titan were developed as ICBM's, but are also the mainstays of the Space Program until the Saturn boosters become available.

1. Atlas, among other things, launched the earth-orbiting capsules of the four astronauts in Project Mercury.
2. Titan will be used for our second step in space, Project Gemini, which will put two men into earth orbit for periods of a week or more. There they will bring two spacecraft together and join in space -- essential experience for Project Apollo.

3. The two-stage Saturn I-B will be used for earth orbital development flights of the Apollo spacecraft and for training the lunar crews.

4. Perhaps the most urgent engineering challenge of Apollo is the development of the three-stage Saturn V -- a launch vehicle powerful enough to send the equivalent weight of two freight cars hurtling across 230,000 miles of space to a lunar landing.

The J-2 and F-1 engines for the various stages of the Saturn V vehicle will provide power for the Apollo launch vehicles. These engines are huge; they dwarf a man.

Each of these engines must undergo a successful test firing before it becomes part of a space vehicle. This requires construction of a carefully instrumented and controlled test stand, where the engine can be anchored down, and then turned on full force for a static test. (Figure 2) One indication of the forces at work is that this F-1 engine consumes one ton of kerosene (RP-1) and two tons of liquid oxygen (LOX) per second.

Development of these engines and launch vehicles is the responsibility of Dr. von Braun who heads MSFC at Huntsville. There a number of stands have been built for static test of the various engines. Such a stand combines the problems of high stress and vibration, tremendous thrust, torrents of incandescent gases, and elaborate plumbing for various exotic rocket propellants, all in a single massive structure.

Some stands are for individual engines, but the stand in Figure 3 is now under construction at MSFC to test the cluster of five F-1 engines which will make up the booster stage of the Saturn V launch vehicle. Each of the concrete pylons is 48 ft square at the base.

The growing power of engines and stages which must be test fired called for a new test center at an isolated location, but still with access by water to the Gulf. This led to the decision to build the MTF in Southern Mississippi, about 40 miles northeast of New Orleans.

Construction began on this unusual facility a year ago. Initial work includes four stands for test firings of the Saturn V boosters. (Figure 4) The test area will be serviced by 15 miles of canals so that the large rocket stages can come in by barge and be unloaded directly on the static test stands.

Figure 5 shows the concept of one of the 400 ft dual position stands to be built, showing the Saturn booster being transferred from its barge. The five F-1 engines of this booster generate the combined power of one-half million automobiles. Imagine the problems of anchoring down something so powerful it can lift 4,000 tons off the earth or put 120 tons into orbit! This work at Huntsville and Mississippi is being carried out by our District Engineer at Mobile.

Meanwhile, in Eastern Texas near Houston, our Fort Worth District is building the ultra-modern Manned Spacecraft Center.

Work on the Center is moving along on schedule, with many of its buildings completed, and several of the more technical facilities under construction. Still others are under design. This Center is a combination operation, research, testing, and training facility which will manage the development of the space ships, train the astronauts, and control their flights. A number of these Space Age buildings are quite unusual.

They include an enormous centrifuge which will subject several astronauts at a time to the high gravity stresses of launch and re-entry under vacuum conditions. (Figure 6)

Another is the largest environmental test facility in the U. S. This will simulate on earth the conditions of space and the lunar surface; such as intense light or total darkness, and temperature extremes from 260°F above zero to 315° below, all in a near vacuum. In fact, the air pressure in these chambers would have to be increased 76 million times to bring it up to the atmospheric pressure at sea level. Obviously the spacecraft, the astronauts and their protective space suits will be well tested under such conditions. (Figure 7)

Another special facility is the Integrated Mission Control Center which will direct future Gemini and Apollo space flight missions and maintain contact with our astronauts far out in space.

Basic facilities have been completed at the Manned Spacecraft Center and about 3,000 NASA personnel have moved into their facilities. However, construction of laboratory and other research type buildings will continue over the next two years.

When the astronauts have been trained and their space ships perfected at Houston, and the mighty boosters have been successfully tested in Mississippi, all will move over here to Cape Kennedy to be assembled into the space vehicles for launch on their long journey.

A harbor was built here on the southern end of the Cape to support the test programs. We plan to double the size of this harbor in the near future to accommodate the increased traffic expected.

Since 1950, over a billion dollars has been spent at Cape Kennedy to convert an abandoned World War II Naval Air Station into the Free World's largest rocket test center. This is a portion of the Cape as it looks today. (Figure 9.)

In the foreground is Complex 37, for launching the Saturn I. So far it is the largest launch complex in the Free World -- and the first built for peaceful use of space. From this complex in late January, a Saturn I launched the largest payload -- about 19 tons -- ever put into orbit. Behind it is Complex 34, which has launched the first four Saturn flights to date. Beyond are the Titan I and Atlas stands, later modified for Mercury, Gemini, and other space programs.

Before we send a man to the moon, a tremendous program is still necessary at Cape Kennedy. Here, over the coming years, more than a billion dollars will be spent to prepare this historic site for its even more historic role. All of this construction work is now being carried out by our newly organized Canaveral District.

The map in Figure 10 shows the present Cape Kennedy - Merritt Island area. To the North, is the new area on Merritt Island which has been acquired for NASA. This $55 million real estate program has expanded the test center to six times its present size. Here, an area largely made up of rattlesnake and alligator infested swamps, with scattered orange groves and a few small palm-fringed housing developments, is being converted into the "Gateway to the Moon."

Here we are building the science-fiction type structures of the Space Age, starting with two giant complexes.

The first is for the Titan III, the Air Force's largest entry in the current space program. This will launch the MOL, Manned Orbiting Laboratory, into an earth orbit to enable astronauts to conduct observations and military experiments in near space just beyond the earth's atmosphere. Titan III will consist of a modified Titan II ICBM with two solid fuel motors, of a million pounds of thrust each, strapped on each side, to carry over 12 tons into earth orbit. Construction of this project is well underway. This is the launch pad, as it will look when completed. (Figure 11.)

Large as this facility is, it will be dwarfed by the new NASA Complex 39 that will launch the Saturn V and its three-man Apollo spacecraft to the moon.

The various stages and components will come by barge to the unloading basin, in the lower center, then move into this Vertical Assembly Building. Here in a controlled environment, the Saturn V will be assembled and erected on its launch rack.

Figure 12 gives a close-up of the $100 million assembly area, showing the Barge Unloading Facility, the Launch Control Center, and towering over them, the huge Vertical Assembly Building.

At the left a launch vehicle is seen all checked out, with its spacecraft in place, ready to move on to its pad several miles away. In the distance a launching is underway. Inside this VAB the unobstructed headroom space is high enough to clear a 45-story skyscraper, yet it must be hurricane proof, water tight, and able to ride out the worst storms ever recorded in this area.

Figure 13 gives another view of the VAB. Nothing like it has ever been built. Its doors are 456 feet high. Here the Apollo vehicle is shown moving out in flight attitude on its 2,000 ton launch platform, its 400' umbilical tower in place, all mounted on a giant crawler-transporter -- the world's largest land vehicle. This vehicle and its load weigh about 17 million pounds. The launch platform is half the size of a football field. It is difficult to see the men, they're so small in comparison.

The VAB will be connected to three launch pads like this by eight miles of crawlerways raised above the swampy terrain.

Figure 14 shows the launcher-umbilical tower, the Saturn and the arming tower in place on the supporting piers at launch site, and the crawler-transporter withdrawn. Here the fueling begins -- 3,750 tons of kerosene & LOX. The arming tower will be moved back to its parking area prior to launching man on his greatest adventure.

However, our discussion of ground support for man in space must not be limited just to terrestrial facilities. We must also consider the requirements for extra-terrestrial support facilities.

Studies are now underway involving systems to support initial surface reconnaissance operations in the immediate post-Apollo period. Today the lunar base is not an approved program, but such a base appears to be an essential part of our national space program in the next decade. Facilities of some type will be required to support the manned surface activities needed for exploration of the moon. If exploitation of lunar resources becomes feasible, more extensive facilities will be required. We in the Corps of Engineers are interested in this effort as an extension of our present efforts in support of Apollo.

First, let's look at the location for these facilities. Figure 15 shows the moon as it would appear to an observer looking through a telescope. However, the engineers and scientists concerned with selecting lunar landing sites, base locations exploration routes, etc., need more information than is shown in this photograph. One of the more recent efforts to provide this information is shown in Figure 16, which is a lunar topographic map at a scale of 1:2,500,000 produced by the Army Map Service, an agency of the Corps of Engineers. This map shows the central portion of the visible side of the moon. The initial manned landings will probably take place in an equatorial band between ± 10° latitude.

The contour interval on this map is 1000 m. with supplementary contours at 500 m. Unfortunately, this is about the best product that can be made at this time with photographs taken from earth-based telescopes. Future photographic information obtained from Ranger, Surveyor and Lunar Orbiter flights should yield much more reliable lunar topographical data. Information which may be obtained from the Surveyor spacecraft lunar landing should also enable us to improve the photogeologic maps of the moon which were first prepared in 1960.

Available information concerning the lunar surface, lunar materials, and the lunar environment is incomplete, subject to interpretation, and controversial. Therefore, until direct evidence is obtained, lunar conditions can be described only in general terms. Large-scale surface features can be divided into two categories: highlands and lowlands. The highlands, or continents, are extremely rough, containing mountain ranges, ridges, rills, faults, and a large density of steep-wall, circular craters ranging up to 180 miles in diameter. Slopes vary generally between 0° and 15° with some of the smaller craters exhibiting interior slopes up to 35°. The lowlands, "lunar seas" or maria as they are more commonly called, are generally featureless having the appearance of large, flat plains. Slopes reach a maximum of 2° or 3°. Craters up to about 50 miles in diameter occur at random in the maria, but at a much lower density than in the continents. The entire surface is covered probably by a low density, highly porous dust and rock froth. This surface layer should be no more than 1 to 6 inches thick in most places, but locally may accumulate to several feet.

The main characteristics of the lunar environment which are of interest from an engineering viewpoint are:

a. Atmosphere -- Essentially non-existent with a density of less than 10^{-12} relative to the earth's atmosphere at sea level.

b. Temperature -- Varies from a maximum surface temperature of 390 K (117 C) at the subsolar point to a nocturnal value of 105 K (-153 C). Changes in surface temperature occur very rapidly with the transition from day to night or night to day.

c. Radiation - Two principal types of penetrating radiation are of interest: solar and galactic cosmic. Flux rates and solar flare events levels have been established within a factor of about two.

d. Meteoroid Impacts - Size and flux of these high-velocity particles have been estimated, but there is no empirical information on which to assess the reliability of these estimates.

e. Gravitational Attraction - Value at the lunar surface is one-sixth of that at the earth's surface, or 5.31 ft/sec^2.

The establishment of any type of facility in such a hostile environment will be a very difficult task. Due to the current state-of-the-art in many fields, it is impossible to determine just how difficult such a task may be. For example, consider the design of a personnel shelter. Since there is no atmosphere on the moon, personnel shelters must be sealed and pressurized to an earth-like atmosphere. Before such shelters can be designed, we must know how materials will behave under high vacuum and extreme temperature ranges. We must insure that meteoroid particles will not puncture the pressure vessel; therefore, we must have information about the effects of hypervelocity impacts on materials. The shelter must also provide protection from ionizing radiation.

As in all lunar systems, the shelter must be designed for extremely high reliability. Since NASA estimates a cost of about $5,000 to transport a pound of material to the moon, our guide for selection of materials and systems stresses lightness and reliability rather than the cost.

The Army Corps of Engineers has been considering seriously the problems of lunar construction for several years. As the primary governmental construction agency, the Corps of Engineers must be prepared to undertake whatever construction mission may be assigned, regardless of its nature or location. One recent effort was a lunar construction research study completed last year.

The purpose of this study, conducted at the request of NASA, was to define the research and development effort required to provide the United States with a lunar construction capability beginning in 1968. Comprehensive studies were made of several engineering areas considered to be of major importance in developing a lunar construction capability. These studies identified many of the problem areas in the design and construction of lunar base facilities and outlined appropriate research and development tasks. These tasks were then integrated into a time-phased plan of action for achieving a lunar construction capability on a schedule compatible with NASA thinking for manned lunar exploration.

Results of studies in some of the problem areas are summarized briefly below:

a. Construction Materials - It appears that current materials technology will meet the needs of most lunar construction design problems. Improvements in present design concepts should result, and answers to yet unsolved materials problems can be anticipated. However, existing data on materials for space applications have been, for the most part, produced and assembled in isolated and uncoordinated activities. This information needs to be assembled, correlated and placed in a form for application to lunar construction problems. Research needs to be initiated for some materials applications such as studies in thermal balance and investigations of lubricants, lubricating techniques, and heavy duty bearings. Testing standards need to be developed for evaluating materials behavior under environmental conditions.

b. Structural Systems - Structural systems will include shelters, maintenance structures, chemical storage containers, communication antennas, towers, and smaller structures such as power and life support modules. Whereas extreme reliability is essential in every aspect of lunar construction design, the high delivery costs demand the optimum in exact analysis and design. Reduced gravity and the absence of certain external loads common in terrestrial design will help in obtaining high performance, lightweight design. Reliable astrophysical data and procedures are needed for appropriate analysis by computers for design optimization.

c. Construction Tools, Equipment and Techniques - The construction tasks to be performed on the moon may be similar to terrestrial construction in many respects. However, many of the tools, equipment and techniques to be employed in these lunar tasks will probably not resemble their terrestrial counterparts. Restraints imposed by the lunar suit, the phenomena of constant mass and reduced gravity, and other environmental conditions will introduce unprecedented design parameters in the development of equipment and procedures. Manual procedures and equipment will be required for unloading spacecraft and subsequent transportation and handling of payloads. Some surface modification may be required. Excavation for placement and protection of base facilities may be the most difficult and critical of all construction operations. Research and development are needed for surveying and mapping instruments, special anchoring devices, and a family of hand tools and other multi-purpose devices for construction, operation and maintenance of the base.

d. Electric Power. Considerable work has been done in developing electric power and equipment for use in spacecraft and satellites, but no great amount of research has been directed to the problem of developing the sizes and types of electrical equipment required in a lunar base. Fuel cells, batteries and solar cells are promising sources for initial power demands of lunar exploration systems. However, the power system supporting any lunar base should include a prime power source with decentralized and mobile power units which may be integrated into an expandable base complex. Considering the probable power requirements for a permanent-type lunar base, nuclear power is the optimum choice as prime power source; however, solar systems may well be used for certain tasks. Early definition of power requirements is essential because of the long time and major effort required to develop a suitable plant.

The Corps' study concluded that a lunar construction capability does not exist today, but it must be developed if landings on the moon are to be exploited. A three-part program was recommended for achieving this capability. First, a comprehensive lunar base study to define detailed concepts and planning for lunar bases. This master plan will establish base requirements and will then identify, define and schedule all significant construction and engineering capabilities required. It will also define performance requirements of related man-systems and space transport vehicles.

Second, the initiation of a technical engineering development program. This integrated program was based on the series of studies discussed previously in each of the major engineering areas involved in lunar construction.

The third part of the recommended program concerned terrestrial facilities needed in the development program. Two of the facilities are of special interest.

The Operations and Test Facility shown in Figure 17 is a large "field house" type of facility which provides a lower-order simulation of certain lunar phenomena under ambient earth atmospheric pressure. The interior of the light-tight building will be finished in black to simulate deep space, and lighting conditions on the moon will be simulated. A portion of the floor will be covered with a lunar soil simulant. In this facility, real-time studies, tests, and evaluations can be conducted on facilities, equipment, and systems concepts. It will also be used for testing and training of personnel. This facility will provide an essential bridge between the capabilities provided by natural terrestrial field sites and the costlier capabilities of a high-order simulation facility.

In the final analysis, however, sufficient reliability cannot be achieved until the combined performance of integrated man-machine-materials systems is demonstrated in the highest level of environmental simulation that is economically possible. For a lunar construction research program it is essential that the simulated lunar environment include the lunar surface for testing of equipment, base components and personnel performance on or within the simulated surface material. A review of existing and planned facilities indicated that none would achieve this capability. Therefore, performance specifications and preliminary plans were developed for such a Lunar Environmental Simulator, which is shown in Figure 18.

The chamber would be 65 feet in diameter and approximately 91 feet high. Lunar surface material simulant can be placed in a container 55 feet in diameter and the depth can be varied from 3.5 to 17 feet. An off-axis xenon-lamp system would simulate solar radiation over an area 20 feet in diameter. Intensity of illumination could be varied to simulate lunar-day and lunar-night operations. The vacuum system would include mechanical roughing pumps, diffusion pumps, and cryopumping panels of gaseous helium. The system is designed to maintain test pressures at or below 10^{-5} torr.

Last July, NASA began a study program to outline a tentative lunar base concept. The concept being studied envisions a flexible, building-block approach for the lunar base. The simplest base will consist of a single module. Other modules will be added for larger bases or for extended missions. These modules are described below.

a. Personnel Shelter. The basic module of the lunar base system will be a shelter designed to house several men. It will have integral life support, power and communication equipment, and could function virtually alone as the principal element of a small temporary outpost. In larger installations, separate subsystems will provide additional capabilities for life support, power and communications, and the equipment installed in the shelters may be retained primarily as standby for emergency use. In emergencies the shelter should be able to house twice its normal complement of personnel.

b. Life Support. All but the most temporary outposts will be served by a central life support subsystem. The subsystem will consist of several modules, the number being dependent on the population of the base and the desired degree of logistic independence. Installations in an early phase of development or small installations for short missions will be served by modules which perform only the most essential functions. In larger, more permanent installations the basic life support modules will be supplemented by regenerative equipment to reduce the need for resupply. The ultimate base complex may include additional modules to generate life support materials from lunar resources.

c. Power. For other than the smallest installations, nuclear power plants will serve as the basic source of energy. A single plant may be used for short missions and during the early phases of base development, but multiple plants of standard design will be the usual source of power.

d. Vehicular Fuel. The vehicles and equipment required for construction and operation of the base will require substantial amounts of energy. This energy will probably be in the form of chemical fuels. In the smallest installation the fuel subsystems will consist of modules which only store and dispense this fuel. To reduce the logistic burden to a minimum, larger installations will be equipped with additional modules which regenerate fuel using energy drawn from the nuclear power plants. In the ultimate base, additional modules may make up fuel system losses by generating fuel materials from lunar resources.

e. Communications and Control. The basic module of this subsystem will provide local and Moon-Earth communications. In addition, it will serve as a navigation aid to spacecraft. Other equipment may be required to provide long range lunar communications and to permit personnel on earth to assume some of the burden of operating the power, life support and fuel systems.

f. Maintenance. Maintenance modules will provide installations of various sizes the capability to maintain vehicles and installed equipment.

g. Vehicles and Mobile Equipment. Construction and operation of a lunar base will require vehicles for materials handling, surface modification, transportation and reconnaissance. Multiple-purpose vehicles may be practicable.

h. Mission Support Equipment. Specialized equipment to support specific scientific missions will be needed.

The study program, now referred to as a study of Lunar Exploration Systems for Apollo (LESA), is divided into three phases. Phase I is an initial concept study to define a reference system which serves to guide the other studies toward a common objective. Phase II is a number of studies of the various subsystems which examine in detail the engineering and operational implications of the base concept. Phase III is an integrated conceptual design study which will re-examine the initial reference concept and incorporate changes indicated by the subsequent subsystems investigations.

The initial concept study was completed by the Boeing Company last December. Four phases were selected in this study to represent activity levels as the lunar base evolves from a small temporary base to a more permanent base which will accommodate more persons for longer durations and increased mission activity. The four base models provide logistic support for 3 men for 3 months; 6 men for 6 months; 12 men for 12 months; and 18 men for 24 months or longer.

The shelter module shown in Figure 19 is designed to be a basic, self-sufficient, support nucleus. All support subsystem modules required in the first base are an integral part of the shelter module. The 25,000 pound payload capacity of the Saturn V transport system also permits a roving vehicle to be carried in the lower part of the basic module.

The interior volume of the aluminum shelter provides sufficient space for six men. Thus, when the base expands to the six-man crew level of Base Model 2, all personnel can still be accommodated in one shelter module. However, additional radiation and meteroroid shielding is required for the longer duration of Base Model 2. The shelter concept provides a caisson as an expandable outer shell of the module. This caisson is expanded about two feet and filled with lunar soil to provide the necessary shielding. Lunar soil is also placed on top of the shelter. Additional life support supplies, power units, fuel modules, and other supplies and equipment required in Base Model 2 will be delivered to the lunar surface as a single Saturn V logistic payload.

Base Model 3 is established by delivering a second shelter module to the lunar surface; positioning it next to the first module; and connecting the two modules. Additional equipment and supplies are delivered as separate payloads. Finally, another shelter module is delivered, positioned and connected to form Base Model 4 shown in Figure 20. Also shown in this figure are the logistic support payloads used to deliver additional supplies.

Most of the comprehensive subsystem studies in Phase II of the LESA program are just being initiated now. However, at the beginning of the program it was believed that two subsystems -- the nuclear power plant and an engine and regenerative fuel system -- would probably most strongly influence the overall base design and might present the most serious development problems. Thus, these studies being done by the Westinghouse Electric Corporation under supervision of the Corps of Engineers, were started last September.

The objective of the multi-purpose engine-fuel system study is to define a concept for a complete system which will provide the power required for the initial shelter module, portable applications, and lunar surface vehicles.

During the early phases of manned lunar operations, the system will be a simple lunar refueling system which will be completely dependent upon resupply from Earth. As the lunar base grows in complexity and power demands increase, base requirements will be furnished from a nuclear power plant. When this occurs, the engine-fuel system can be supplied with fuel regenerated from the engine reaction products by a processing plant operated from this prime nuclear plant. Thus, dependence on earth resupply will be reduced. Ultimately, it may be possible to achieve further independence from earth by using natural lunar materials if their use as fuels proves feasible.

In the engine-fuel system study, possible power systems were evaluated in terms of engineering, techno-economical, and qualitative considerations. When these factors were combined, a comparison of the possible systems could be made for each application. Five basic sources of energy were investigated: solar, thermochemical, electrochemical, cryogenic, and radioactivity. Both static and dynamic engines were evaluated. The recommended systems resulting from this study which has just been, completed, can be described in terms of the principal applications of the multipurpose engine-fuel system.

A solar cell and fuel cell combination is used for the shelter power system to meet the moderate requirements of Base Models 1 and 2. Since there is no atmosphere to absorb solar energy, the sun provides 130 watts of energy per square foot on the lunar surface. Maximum use is made of this free energy for the relatively constant demands in the shelter module. A solar photovoltaic engine composed of silicon cells produces direct current power from direct conversion of sunlight to electricity. The solar engine consists of a flat array of inter-connected cells supported by a framework which is oriented with respect to the solar flux by means of a simple drive mechanism.

For the lunar-night power supply, a hydrogen-oxygen fuel cell is used. This type of fuel cell has several attractive advantages. One is the modular design inherent in the fuel cell itself. This means that power can be provided in as small a block as necessary or can be extrapolated to meet requirements while retaining good fuel characteristics. Another advantage in using hydrogen and oxygen is that the product of their combustion is water. This water can be used to meet life support requirements or it may be regenerated into the original fuels with an efficiency that makes regeneration in the later bases an attractive part of the energy plan.

In moving applications, it was considered impractical to use a solar engine to reduce logistic weight because of the difficulties of orienting the solar cell array. Thus, only a fuel cell is recommended for vehicle engine systems. Optimization of engines for vehicular applications is very difficult until specific mission requirements are established. For long operating periods without refueling, it may be necessary to use an isotope-power system which would require biological shielding.

A silver-cadmium battery is the preferred system for portable power requirements of 1 kw or less. For this application, the primary requirements are simplicity, convenience, light engine weight while in use and minimum bulk. The silver-cadmium battery is hermetically sealed and produces a given amount of energy for a given weight. Thus, the power level is flexible, i.e., a large power can be drawn for a short period or vice versa. For the larger portable power supply, the preferred engine is the fuel cell.

The nuclear power plant study, still underway, is investigating nuclear reactor power system which could meet the major power demands of the larger lunar bases. The initial base concept study indicated that startup of the first nuclear power plant becomes economically appropriate when Base Model 3 is established. The high costs of unscheduled power outage resulting from large costs for transportation of back-up power systems and fuels, and from high costs due to loss of base functions both in terms of dollars and possible complete disaster, place heavy premiums on high plant capacity, long plant life and very high reliability.

Figure 21 shows the nuclear power plant module as it would be delivered by a Saturn V logistic vehicle. Possible over-all plant arrangements and associated modes of packaging range from a relatively low capacity plant with integral shielding (i.e.,brought from earth) and a fixed radiator to a potentially much larger plant with all or much of the shielding accomplished by lunar material (e.g., by burial of the reactor) and a deployable radiator. An integral shield would take up a large portion of the available payload weight. However, such a concept has advantages for the early plants because of their small demand on the limited available construction effort and their insensitivity to lunar surface characteristics.

The concept being developed envisions locating the plant about two miles from the personnel shelters. This separation distance provides sufficient radiation attenuation so that the plant shielding can be reduced to an acceptable payload weight. Studies of electric power transmission from the plant to the base proper have indicated that it is feasible to transmit over this two-mile distance with a transmission system of modest weight. A typical lunar base transmission line would consist of bare conductors, operating at elevated temperature, losing heat essentially only by radiation to space. The conductor would probably consist of several parallel conductors with multiple interconnections so that the transmission system could withstand meteroroid hits and still function.

The nuclear power plant concept currently being investigated in the study is shown in Figure 22. There are reasonably well developed technologies on which to base a design which has a good probability of meeting the program objectives. There is, however, no existing plant which will fit the anticipated requirements of the lunar base. Thus, an aggressive development program is required. The present concept is for an initial plant rating of at least 100 kwe and a growth potential to 300 kwe. The reactor will operate at a high temperature and use an enriched uranium fuel. The power conversion system will be a liquid metal Rankine cycle.

One of the important constraints on engineering a power plant for lunar use rather than terrestrial operation is that heat must be rejected by radiation. Thus, a radiator is required for space electric power generating systems. However, the meteoroids present in the space environment make it necessary to provide sufficient armor for the radiator to prevent meteoroid penetration which would release the coolant. If all the coolant is lost,the plant becomes inoperable. To improve the survivability of the radiator, as well as somewhat reducing the armor requirements, multiple radiator loops are used.

The radiator of the proposed lunar nuclear power plant, as can be seen in Figure 22, is composed of about one hundred closed end tubes. At the base of each tube is a bellows joint with a mechanical hinge that permits deployment of the radiator as shown in Figure 23. In this position the total radiator area is approximately twice that available on the surface of the capsule if the radiator were not deployed.

This two phase, natural circulating, refluxing radiator also takes advantage of the lunar gravitational force. The turbine exhaust vapor goes into a large manifold which contains small boilers attached to the end of each of the radiator tubes. These boilers contain liquid potassium. The vapor from the turbine condenses on the boilers; the liquid metal boils; the metal vapor rises into the tube where it condenses on the surface by radiation; and the liquid then drips back into the boiler.

Concepts for other subsystems will be defined in detail in the other Phase II studies of the LESA program. When the integrated concept study is completed in Phase III, NASA expects to have a sound and comprehensive plan for a lunar base system.

CONCLUSION

This paper has covered briefly some of the broad engineering and construction support which the Army Corps of Engineers is putting behind this nation's manned space program. The first phase of the journey to the moon will be right here on the ground.

Our future successes in space are dependent on the nation-wide effort which NASA directs and spearheads, but certainly an important element of this is the accelerating program to provide the complex ground support facilities required throughout the effort. This program involves design of highly sophisticated structures, much of it to criteria which transcend the state of the art; development and procurement of special and unusual components which tax the capabilities of the industry; and construction to tight schedules and exacting standards which are foreign to the industry.

The next phase of our manned lunar landing program will be the support facilities required on the moon. Providing these facilities will require solving unprecedented engineering problems imposed by delivery systems limitations and the harsh environment about which there is little information. A significant development program is needed to develop the equipment, techniques and reliability necessary for this difficult task. Such a program must be started now if we are to have an orderly program rather than a wasteful, crash program later -- wasteful in terms of personnel, materials and funds.

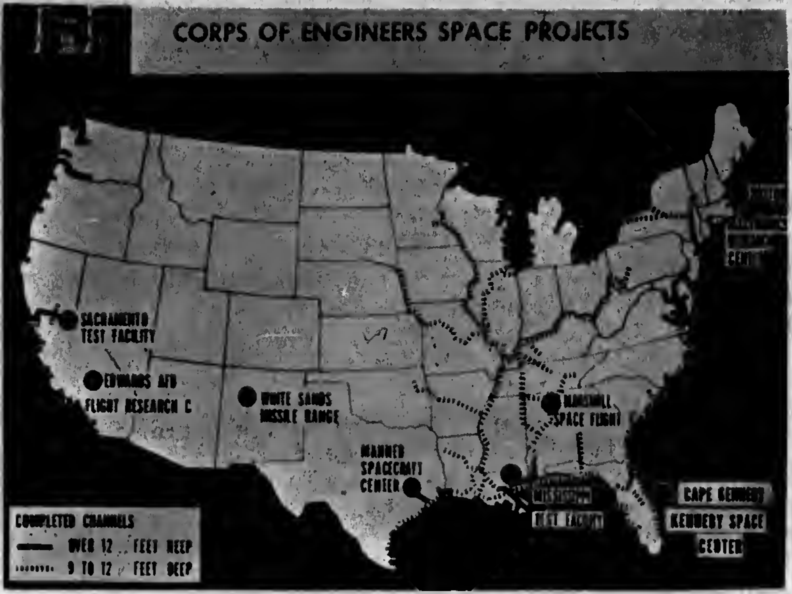


Figure 1.

Figure 2. F-1 Engine Undergoing Static-Test on Stand Constructed by Corps of Engineers

Figure 3.

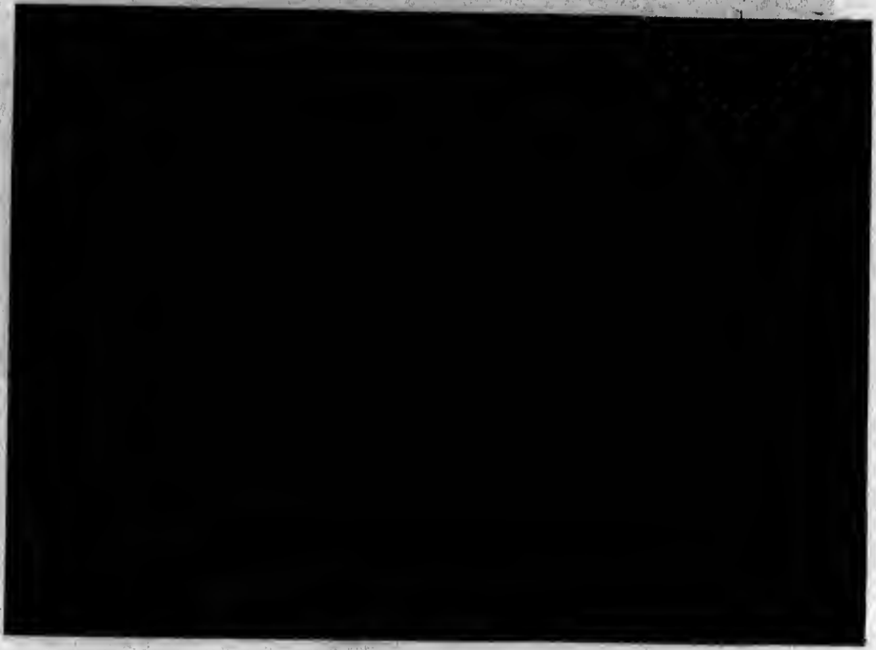


Figure 4.

Figure 5. Dual position stand at Mississippi Test Facility under construction.

Figure 6. Huge Centrifuge being constructed at the Manned Spacecraft Center near Houston, Texas.

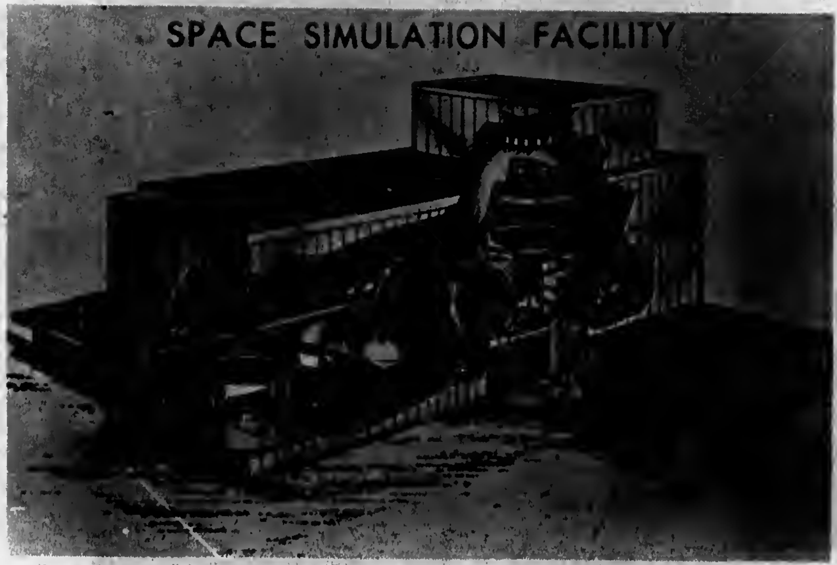


Figure 7. Space chambers going up at the Manned Spacecraft Center.

Figure 8. Artist's concept of completed Manned Spacecraft Center.

Figure 9. Cape Kennedy as it looks today.

Figure 10. Map of Cape Kennedy and adjacent Merritt Island launch area now under development by Corps of Engineers.

Figure 11. Titan III launch pad being constructed for the Air Force.

Figure 12. Complex 39, under construction for NASA, will become the Nation's Moonport.

Figure 13. Close-up view of the Vertical Assembly Building and Saturn V on crawler-transporter.

Figure 14. Saturn V shown lifting off on flight to Moon.

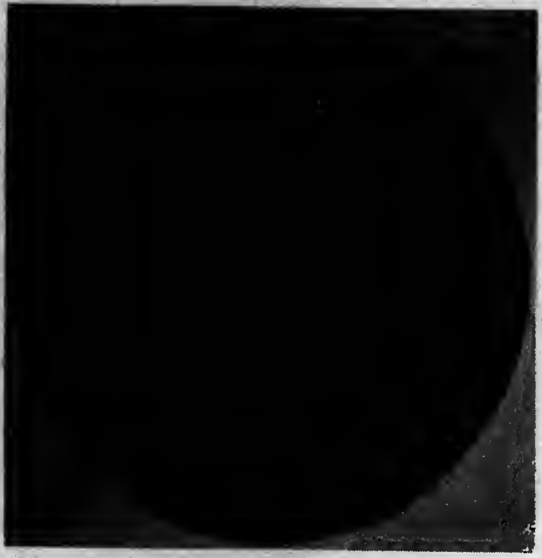


Figure 15. The Moon as seen through a telescope.

Figure 16. Topographic map of moon shows area where initial manned lunar landing will be made.

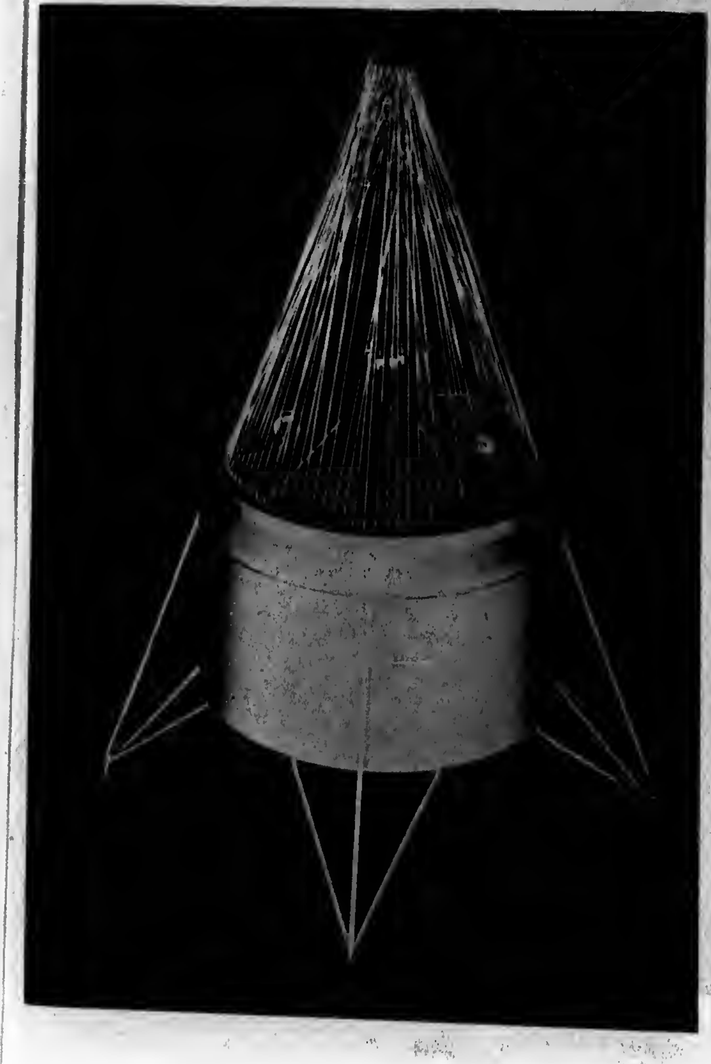
Figure 17. Proposed Operations and Test Facility for simulating certain lunar phenomena.

Figure 18. Environmental simulator needed in a lunar constructi research program.

Figure 19

Figure 20

Figure 22.



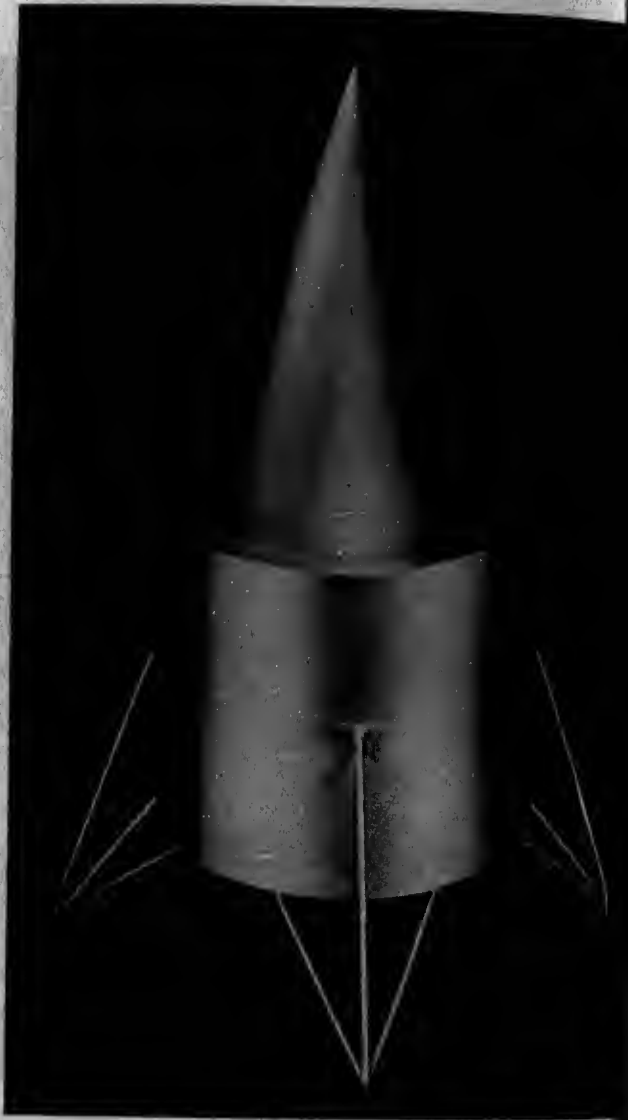


Figure 21.

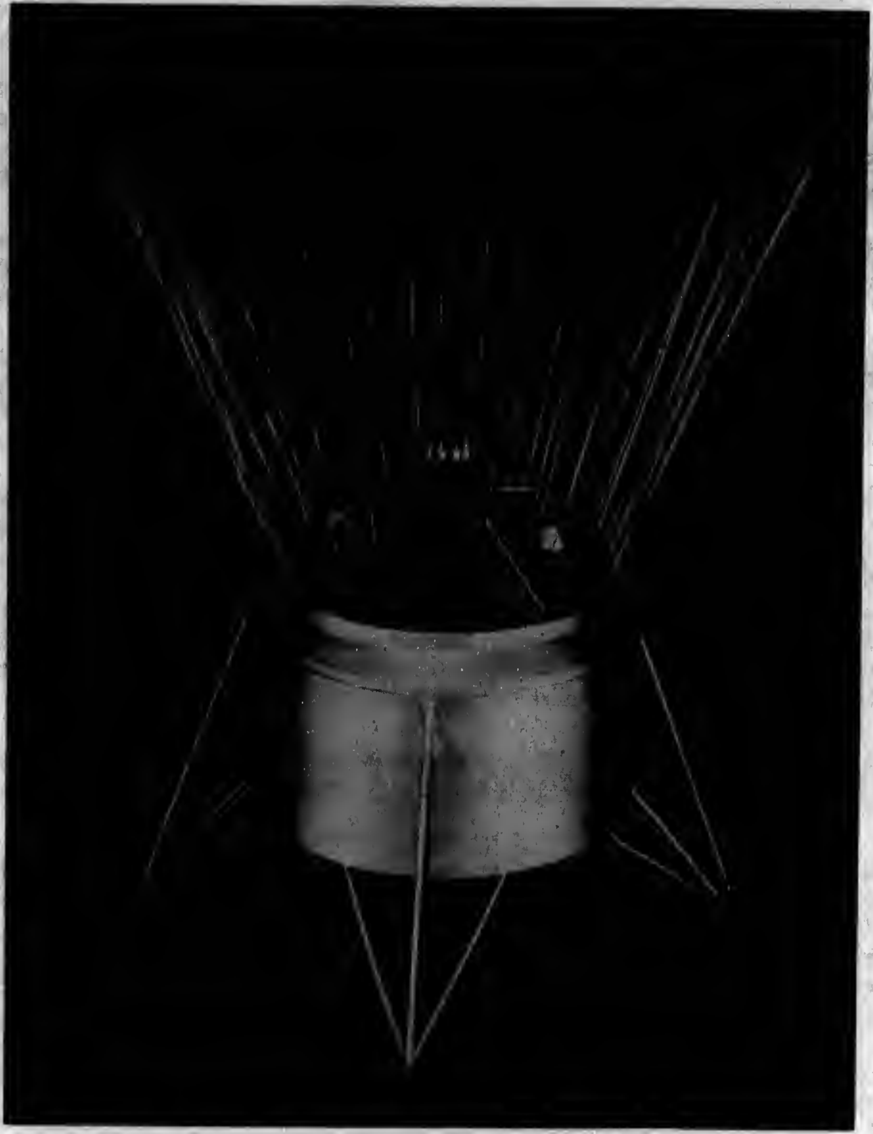


Figure 23.

STRUCTURING THE ENGINEERING ORGANIZATION TO SOLVE THE INFORMATION PROBLEM

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SUMMARY

The handling and use of vast and diversified information in today's science and business, can no longer be thought of as "just a communication problem". It is a management problem. The theme presented here is not offered as a panacea for ASTIA or other automated retrieval systems. Rather it is directed toward information management in the functional technical organization.

A Research or Engineering Manager wants to be sure that the organization receives pertinent information and that is is immediately channeled to those who need it.

The consequence of not meeting information needs are a number of costly losses. Existing ideas will be reinvented. Work will be duplicated in different departments or divisions. Failure to capitalize on internally generated ideas.

Individual inefficiencies will result. Not so obvious, is lowered morale, resulting in the loss of good people. Tragically it happens before it is seen.

There are a number of barriers to solving this problem ... psychological, sociological and structural. However, there are a number of approaches to overcoming these barriers. Of the three, the simplest, most effective, and controllable is to change the organization structure by adding a communication function or discipline at the lowest level.

The guidelines for designing the structure are listed. A typical diagram is shown. A pilot plan and implementation is suggested with emphasis on certain key points.

How the new structure benefits the Manager, as well as the professional employee, is clearly shown, and the economic gain to the organization is estimated.

There are many other benefits of this new system.[1] Increased efficiency due to elimination of duplication in searches and storage; cross fertilization of ideas from many sources; central sources for retention of valuable experience is provided; elimination of haphazard communication and non-essential mangerial paper work; reduction of distortion in transmissions of material; and, last, but not least, a truely integrated organization can result.

INTRODUCTION

"Bismark is quoted as saying, "If I want a Minister of Education all I need is a man with a long white beard, but if I want a Chef I need a man of infinite genius". Often we wonder if we are Ministers or Chefs, Managers or Specialists. Are we keeping up with the state of the art? Do we ever dream of tackling another interesting discipline ... even if we have the talent? To do such things requires more depth and breadth than we can individually command ... assistance in finding, assimilating and applying the collective intellect of man to the problems we face. In short, we face today the age old but increasingly complex problem of man in moving down from the trees and up to the moon communication.

Today's technical information is massive. It's growth exponential. There is literally a flood of military technical documents, reports, journals, magazines, proceedings, abstracts, methods, thesis and books. Add to this such internal information as correspondence, reports and data and the staggering total produces a crisis in communicating information. The handling and use of such vast information in today's science and business can no longer be thought of as, "just a communication problem", it is a management problem.

Professor Elton May[2] said, "We have passed beyond the stage of human organization in which effective communication and collaboration were secured by established routines of relationships". We are moving into an adaptive society. Change is the order and necessity for survival.

This paper is not offered as a panacea for ASTIA, STARS or other automated retrieval systems. Rather, the new approach is directed toward adapting and modifying the technical organization's structure to more adequately handle its information needs.

THE ENGINEERING MANAGER'S NEEDS

The Technical Manager of a project wants timely receipt of all necessary pertinent information by his engineers. How can he be sure this is done without doing it himself?

Basically, he must assure two actions which are within his sphere of control: (1) find the relevant information; and, (2) insure its immediate transmission to those who need it.

THE CONSEQUENCES OF NOT MEETING INFORMATION NEEDS

Scientists and Engineers face a situation where they are working on problems for which a solution may already have been found. This reinvention is costly. Quite often commercial projects duplicate the military. In large companies the same work may be repeated at a later date by a different department or even in the same department due to lack of adequate records and communication. This causes embarrassing loss of time.

Quite often valuable ideas are generated in one department but are not shown to top management.

Insufficient inter-department communication results in lost knowledge: a research department fails to pass on its findings to the development group. Such an information monopoly can exist when any one man who knows certain answers does not volunteer his information in solving a problem that is not his direct concern.

Lowered morale is another result of poor communication which can cause a high turnover in personnel.

BARRIERS TO ACCOMPLISHING IMPROVED COMMUNICATIONS

Bhe blocks or barriers to solving this communication's problem are numerous. They are structural, sociological and psychological. The actual organization structure is the formal framework within which each man works. The formal channels should handle the majority of the information. Where they are inadequate, self-made informal channels[1] may tend to develop to supplement them. However, inadequate structure is often not by-passed since, as Managers develop their consciousness of status, their self-interest in maintaining artificial control causes them to block, upward or bilateral, information flow except through them.

A sociological block develops when the individual professional employee develops a boss (status) consciousness which materially stops communications. This occurs particularly where the boss lacks empathy.

Professional workers[5] have vastly different training and experience ... including language concepts and meanings ... and hence psychological blocks may develop among these heterogeneous peers. To compound the problem the inability of the Manager to assess individual information needs, the communications flow.

Another barrier lies in the sheer bulk of the information which must be surveyed to extract all pertinent fact. When all these blocks are interposed in a probabilistic manner, enough of them are simultaneous, to render most communication very inefficient.

APPROACHES TO OVERCOMING THE BARRIERS

There are a number of possible approaches to overcoming these communication barriers. The first which should be considered is special training for the Manager. This could improve conditions but he may have some inherent limitations (such as time and desire) which could seriously hinder improvement.

Another approach is to train the professional worker.[6] The liklihood of success of this approach is less since he lacks the necessary ingredient of control.

Finally, the structure of the organization can be changed. The simplest, most effective, means of doing this is to add a communication function or discipline. A Technical Information Specialist, placed as a staff assistant in first level, is structurally integrated with other organizational elements. This approach has the most promise of eliminating the blocks to communication.

GUIDELINES FOR DESIGNING THE STRUCTURE

The activities and the operating areas of all the technical groups should be analyzed. Special consideration should be given the identification of redumant or overlapping interests.

Next, a study of the inter-department communication paths should be undertaken. In areas where complexity and high information flow or bottlenecks show up, a communication specialist should be added. The job specification should be written to meet the needs indicated by the study. For effectiveness the communication specialist should report directly to the task manager of the group but should have continuous contact with the prime source in the organization for generating external information. The line Managers should be given some training on how to use the Information Specialist. Frequent formal meetings should be set up for contact between all Information Specialists. This is necessary to prevent the task manager from using the specialists to just keeping him informed, defeating the broader information and communications needs of the organization.

A SAMPLE OF THE STRUCTURE

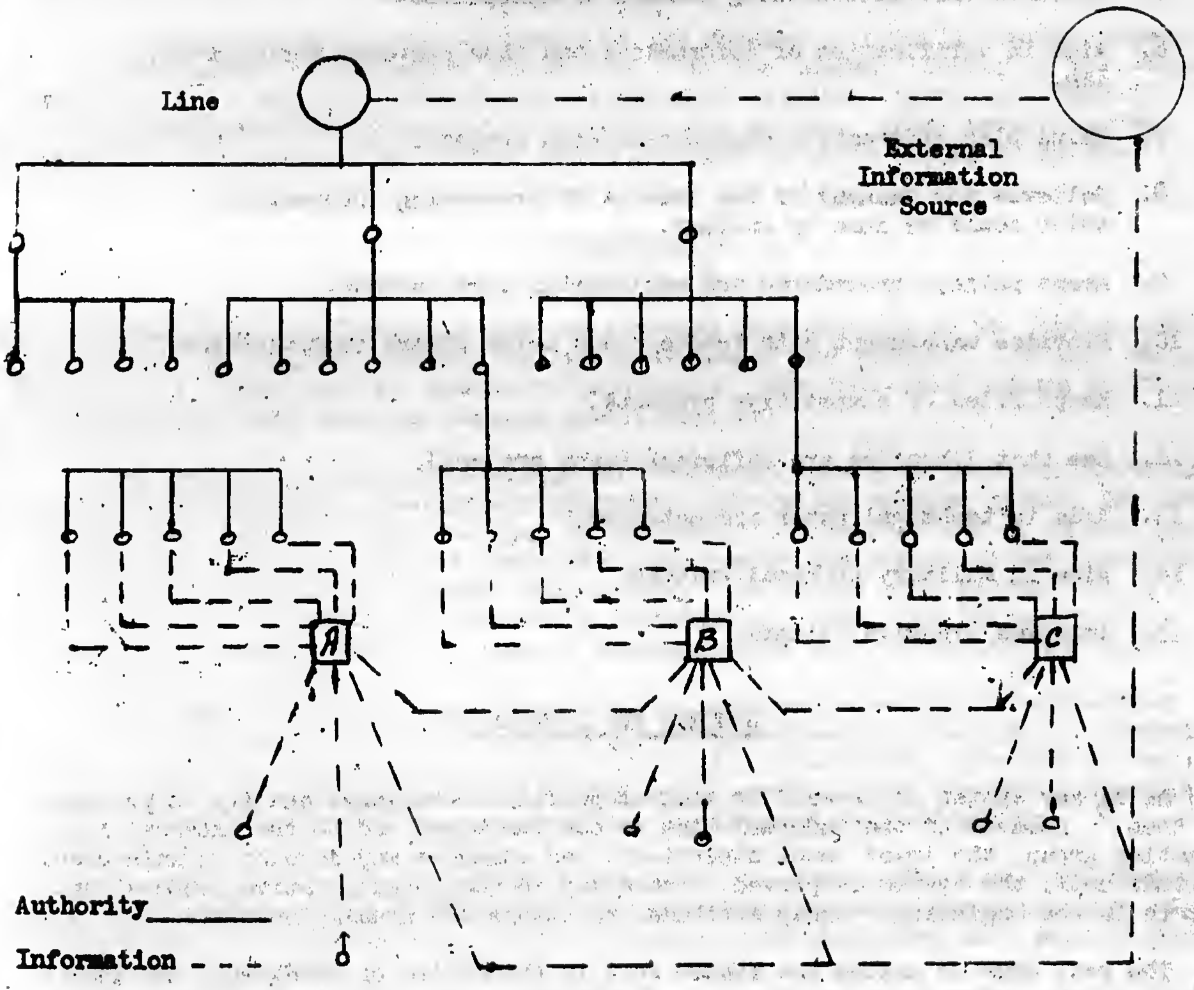

FIGURE 1

WHAT THIS NEW STRUCTURE WILL DO

The new structure (Figure 1) will overcome many of the Communication barriers. In a positive way it will accomplish many of the following needs:

1. Assists individuals who become lax in keeping up with the state of the art.
2. Aids members in laying out searches and preventing duplication.
3. Integrates knowledge of other groups and gains their co-operation.
4. Abstracts articles common to all.
5. Prevents information being hoarded or monopolized.
6. Aids in verification of information and thus prevents verification lag.
7. Helps plan information channels on each project.
8. Relieves the Manager of the details of processing information which could be done by another.
9. Keeps policy, procedures and engineering logs current.
10. Provides management with feedback and helps upward communication.[8]
11. Keeps files of competitive products.
12. See that libraries are efficient (many are not).
13. Aids in technical paper presentation.
14. Aids in military proposal writing.
15. Improves creative climate.

DANGERS AND PITFALLS

As in any system it is well to look at pitfalls envisioned and try to provide for them.[7] Because of the informal ties across the board and to the external information group, the lower level supervisors and managers may develop an unfavorable attitude until the system is proved. There may be some uncooperative professional workers in the beginning. Group attitudes to change are usually negative.

The real keys to making the system work is the desire of management and the ability of the information specialist. If failure occurs it will be in not picking the right man for the job. The importance of selling the whole system can be underestimated, resulting in the loss of integration across the board. Authority or decision making should not in any way be taken away from the manager. Also, the professional worker's personal responsibility to see that he geets needed information should not be passed on to the Information Specialist.

Last, but not least, the structure should not be used as a judgment of individual worth! That is , management by objectives should not be replaced by management through hearsay. Derogatory values passed on by an unguarded tongue, should not be used as a jusgment of any individual. With proper care, and using feedback to check conditions, these dangers can be avoided.

SUGGESTED PILOT PLAN

Make a fully worked out company pilot plan and program on paper. Pick an average technical department for the pilot implementation. Specify all duties of the information specialists and carefully pick a qualified person. A series of Educational Conferences between the Managers and the Specialists in which each one works out his objectives, can be compared with the general theoretical objective stated in the tentative plan. Modify the plan depending upon constraints and resources available.

Be sure to align the self-interest of the Manager to overall company objectives with regard to information objectives.[3] Put the company library in the picture. Set the plan in operation and require an informal feedback reporting system to the information department and manager to correct any faults. After several months experience in one group the system can be expanded to other technical groups.

CONCLUSIONS

The information discipline must be directed as a service to the professional man and not as a special administrative assistant to the Manager. The work should be run on a "management by objective"[3] philosophy. The individual chosen to fill the discipline must have the correct qualifications.

The information discipline should be structured into each technical group and co-ordinated with the organizations' external information service group. The trend in organizations is to have the span of responsibility replace the so called Span of Control. Under the span of responsibility concept the number of men reporting to one individual is much greater than before. Therefore, each Manager can give less time to each man, for transmitting information. Thus, the inclusion of an information function makes the span of responsibility more effective and broader.

Management must be so organized that the problem solver or Professional Employee is most effective.[9] To structure the organization by inserting an information function in each group is a powerful tool for doing this. Starting in one group first, as a pilot plan, then it can be enlarged through the whole organization. A training program is necessary for the Information Professional.

The intangible gains are many, yet are hard to value. The costs are obvious. Top management wants to know that the value achieved is more than the costs. The Field of Information Economics has only recently been developed as a part of the business systems field.

The scientist has been given many elaborate tools to aid him in his work. A careful analysis of his activities indicated his unfilled needs are in the area where human judgment and discrimination are necessary.

Life is brief but art is longer
So the sages say in sooth
Nothing could be worse or wronger
Than to doubt the ancient truth
Endless volumes, larger, fatter
Prove man's intellectual climb
But in essence it's a matter
Just of having lots of time.

Give me half a dozen monkeys
Set them to the lettered keys
And instruct these simeon flunkies
Just to hit them as they please
Lo; the antropoid plebians
Toiling at their careless plan
Would in course of countless aeons
Duplicate an information plan.

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PERSONNEL AND ORGANIZATION DEVELOPMENT IN AN R&D MATRIX-OVERLAY OPERATION

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Summary

Increasingly, R&D companies are finding it desirable to adopt a "matrix-overlay" organization structure to accomplish project goals on a timely and efficient basis. In shifting from a straight functional organization to a combined functional/project form, some key changes must be made in responsibilities of Program Managers and managers of engineering "Capabilities Centers." New standards of performance and means for reward and advancement under the revised organization structure must be developed. Diverse management and social science experience are called upon in planning and implementing the matrix-overlay structure through the development of such standards.

Broadly speaking, there are two organizational structures for R&D work: the functional organization and the project organization. In terms of expected characteristics of the two organizational forms, it can be said that the functional organization encourages growth and development of technical experts while the project organization gets the job done and develops leaders at the working level. There are advantages and disadvantages of each type of organization, but no matter which form is established, the typical research and development operation eventually uses both arrangements with a resulting mix of advantages and disadvantages.

A major dilemma for R&D management is to combine technical people into working groups in such a way as to maximize application of their creative ingenuity while at the same time developing sufficient "end-item" orientation to achieve a practical payoff from the research effort. Organizing researchers into groups based on disciplines protects the integrity of specialized competence. It provides a homogeneous cluster of people which is relatively permanent, with close communication within common areas of interest. In the usual functional organization, however, such groups are inclined to become somewhat compartmentalized, with parochial interests and a continually narrowing purview of the over-all needs of the operation. Some loss of creative output is inevitable under such a setup, particularly where a closely managed, goal oriented R&D program is required.

The straight project management setup, on the other hand, divorces the technical man from his professional group. It places him in an operating environment that is too strictly "project" oriented. It removes him from the technical status relationships, "colleague authority," and scientific reference group identification which are important to development of good technical capability in an organization.

In order to achieve the advantages of both functional and project organization, many R&D organizations have been adopting a "hybrid" type of organizational arrangement, referred to interchangeably as a Matrix, crossbar or overlay organization. In the matrix organization a Program Management group is set up with the purpose of getting the job done by drawing upon the capabilities of functional or line groups, including engineering and support services. In effect, Program Management becomes a quasi-permanent line group. Individual staff members are temporarily assigned to a particular program and upon its completion either return to home base (or "Capabilities Centers") or may remain in Program Management to work on new programs.

It has been found that the organizational changes which are made in implementing the Program Management concept result not only in the expected improvements, but some deleterious side effects as well, particularly insofar as employee motivation and interpersonal relationships are concerned. Thus it is particularly important that a company introducing the Matrix type of organization where a more traditional functional form existed should take steps to maximize the benefits and minimize the negative effects on its people.

Usually, in implementing the Program Management concept in its operation, the company publishes new organization charts and issues directives and bulletins to modify employee performance. Such methods are necessary, but more important, it is the establishment of clear standards of required performance, fair evaluation thereof, and setting up of reward and advancement systems which are responsive to the new needs of the Matrix organization, which help management to obtain the needed engineering acceptance and support.

Changes in Required Performance of Engineering Personnel

By definition, the Matrix organizational setup affects people in all reaches of the operation. However, it is engineering performance which is most crucial to the success of the new organizational approach. For this reason, the required changes in performance under the new management are discussed here primarily from the point of view of the technical element of the operation. A point worth making is that it is not the organizational structure per se that creates a change in the reward and advancement structure, but the accompanying differential performance requirements of higher management that do so. It is only to the extent that new performance requirements with respect to higher level engineering filter down to operating level engineers that it becomes appropriate to develop new standards of operating level engineering performance.

It is presumed that in most organizations operating under the Matrix concept the required performance of operating level engineers (non-supervisory) would be substantially the same as under any previous organization. These personnel continue to report to engineering project managers and, working at a relatively detailed and highly specialized level, do not become involved in changed working relationships as do the higher levels. In their case, standards of effective performance and accompanying rewards remain the same as earlier.

There are several ways in which the Program Management concept has been implemented in organizations, some effective, some much less so. For example, Program Management (or Program "Control" in this case) may be provided minimal authority wherein it is required only to monitor program schedules or provide PERT or Line-of-Balance (LOB) assistance to engineering managers. In this situation, minimal change is found in the program activities of the organization. Little cooperation and minimum results can be expected. On the other hand, Program Management may be provided maximum control and authority over engineering programs and projects with policy guidance and support required from participating engineering line groups. On programs under Program Management, senior engineering managers are seen as heads of support groups or "vendors" to Program Management. In this situation, changing to a Matrix organization (especially from a functional organization) can achieve good end results, but because it involves a more fundamental change in relationships, it usually has a more severe effect upon people and therefore requires maximum attention.

While some specific responsibilities for senior engineering managers drop out under the Matrix concept, and others change in degree of emphasis, one prime responsibility is added by top management requirements. That is the relationship of engineering line groups to the Program Management staff. Engineering groups are now required to be "Capabilities Centers." These Capabilities Centers are foci of capabilities which have a common basis and in which working relationships have been established. They are relatively permanent--the "home base" of the engineering specialist. It is there that he grows and develops professionally. They are "culture pockets" for development of the state-of-the-art within specific technological areas. These Capabilities Centers are required to support Program Management with: 1) skilled engineering talent (knowledgeable in new technologies), 2) facilities, 3) equipment. To make the most effective contribution, the Capabilities Centers must be continuously aware of and interact with present and future needs of Program Management. Clearly, Capabilities Centers must also develop with the aim of staffing for new programs and projects under their own direction. In either case, engineering managers are now, more than ever, to be evaluated by the extent to which they improve that state-of-the-art, infuse new talent, encourage new inventions and concepts or, in other words, maintain high quality Capabilities Centers.

Not all required performance of senior engineering personnel involves capabilities development in support of Program Management. In the typical "mixed" organization, some of these personnel have other programs or projects to manage (not under Program Management control) whereby they acquire or maintain responsibilities held under the previous organization. Standards of performance for these personnel are discussed in a later section.

Program Management's Role in Facilitating Better Engineering Performance

Program Management is seen as a Network of Activity. Its function is overlaid on the basic company structure. Performance required of these staff members relates to planning, coordinating and controlling the numerous activities involved in meeting a program goal. Performance requirements of Program Management should place emphasis on the ways in which it can facilitate the engineering product.

Successful execution of any program under Program Management depends upon successful interaction between the Capabilities Centers and Program Management. The word "interaction" is not an empty euphemism. It is an absolute necessity, inasmuch as Program Management acquires its capability (although it has authority under top management directive) almost entirely from and at the discretion of the Capabilities Centers. In order to perform its job, Program Management must assist engineering Capabilities Centers to perform their jobs. Program Managers must demonstrate their usefulness to engineering and not merely tell engineering how helpful they can be. Program Managers should seek to facilitate engineering in some of the following specific ways:

1. Give each engineering group manager a "manageable" job; one that requires solid engineering capability and is achievable. Provide engineering with all necessary inputs to do the job (e.g., specs., technical manuals, detailed requirements).

2. Develop a full understanding of the problem (possibly through a detailed statement of work and a subsequent bid response from each group) and the necessary capabilities to do the job.

3. Throughout a contract, try to free engineering from administrative paperwork, so objectionable to engineering. One way to achieve this end is to obtain necessary information for reports through meetings with engineering and have the Program Management staff actually write the reports.

4. Aid Manufacturing by performing liaison between them and Engineering by stimulating the timely and accurate receipt of engineering change notices and other drawings and reports. Where clarification of drawings or specs is necessary, Program Managers should provide it to Manufacturing.

5. Coordinate and arrange necessary trips and meetings between engineering and the customer.

6. Track down unauthorized or doubtful charges against engineering case numbers.

7. Ensure and conduct design reviews periodically to assure in-process design adequacy.

In summary, Program Managers must perform two major missions to aid in achieving improved engineering products: They should buy "time" for the engineering project managers to perform their technical work. They should direct engineering as to "what" is required and "when" while allowing engineering to determine "how" it is to be achieved.

Engineering Performance Standards on Programs Under Program Management

On programs under Program Management, the increased time available to engineering management is expected to enable the following emphasis on performance and the accompanying standards of performance:

1. Closer technical supervision. This includes closer cognizance over design decisions and drafting, enabling time for realistic design reviews and drawing reviews. Standards of performance include:

- Fewer engineering changes during R&D including reduced number and extent of changes as a result of design reviews.
- Reduced drafting and reproduction costs on the program.
- Closer adherence to specs during equipment qualification tests.

2. Closer liaison with manufacturing in order to assure compatibility of the engineering product with manufacturing operations. Standards of performance include:

- Minimum number and extent of drawing and spec changes for manufacturing purposes.
- Timeliness of delivery to Manufacturing of an engineering product in order to meet delivery schedules.
- Measured ability of engineering products to meet specifications over the full range of environmental tests.
- Decreases in scrappage (waste) costs which are incurred as a result of ordering against continually changing or inaccurate drawings.

3. Close attention to standardizing equipment circuit and component design. Standards of performance are evaluations as to the extent of engineering adherence to corporate standards (e.g., microminiaturization). Much money and time are needlessly lost in unnecessary replication of such things as flip-flop circuits and power supply design. Standards of performance include cost savings obtained through greater standardization in the following specific areas:

- Through placement of larger orders wherein cost advantages are thereby obtained.

- Through reduction in new tooling. Existing tooling can be utilized to a greater degree.

- Through reduced costs of training new operators.

- Through experience in producing similar products well.

- Reduction in rework or scrappage which in turn results in greater availability of capital equipment.

4. Greater opportunities for recruitment and development of high calibre engineers. Specific measures of performance of these responsibilities can be the following:

- Greater number and quality of high talent and advanced degree engineers.

- Greater number and quality of published articles.

- Greater number of state-of-the-art patents.

- Greater number of valid promotions.

5. Greater order input and billings. Engineering is at least partly responsible for order input and billings. Engineering performance is in large measure a factor in the amount of new and add-on business acquired. Three specific standards of performance are:

- Greater Yield rate (# of Awards/Quote).

- Reduced Expense-of-Capture rate (Expense/Award).

- Increased Percentage share of new and desired markets.

6. Greater recognition of engineering capabilities by outside professional groups and organizations, other corporate divisions and internal groups. Specific measures of performance include:

- Greater frequency of requests for and participation on key technical committees and at technical seminars.

- Greater numbers of requests for and contributions to other company elements.

- Greater numbers of requests for and contributions to other internal corporate groups.

It will be noted that these performance requirements place emphasis on technical excellence for engineering personnel, an emphasis which can easily be overlooked in this age of "massive engineering." The Matrix organization, in effect, makes it possible for the technical man to be a better technical man rather than be forced reluctantly into the mold of a manager.

Engineering Performance and Standards of Performance that are not Affected by Program Management

Since the typical organization applying the Program Management concept is "hybrid" (that is, the larger contracts are under Program Management, smaller ones are managed directly by engineering project managers), not all engineers can divest themselves of management responsibilities; nor are they likely to desire to do so. Those engineering managers who manage contracts not under Program Management will have responsibilities in addition to those mentioned above, with accompanying performance standards as follows:

- Achieve order input at projected $ levels.
- Achieve billings at projected $ levels.
- Meet delivery schedules on time and within costs.
- Maintain customer satisfaction through reduced frequency of complaints.
- Assure proper profit margin.
- Enter new markets in accordance with Corporate Business Plans.
- Administer General Development funds

  a) to achieve technical objectives within costs,
  b) to achieve short and long range payoff.
- Maintain low indirect/direct ratios.

These performance requirements are both business-oriented and technically-oriented responsibilities. They require the proper mix of management and engineering competence.

Areas of Overlapping Performance and Standards of Performance between Engineering and Program Management

To complete the picture, it should be pointed out that certain responsibilities must necessarily be shared by engineering management and Program Management. Standards of performance for these areas include:

- Meeting delivery schedules.
- Operating within budgets.
- Assuring customer satisfaction.
- Sales.

While the above standards are primarily required of Program Management or Sales (Marketing), it would be unrealistic to remove all responsibility and accountability for these standards of performance from engineering Capabilities Centers. We take specific note that interaction and cooperation between Program Management and engineering Capabilities Centers are a sine qua non toward satisfactory achievement of the above standards of performance.

Opportunities for Growth and Advancement for Engineering Personnel under the Program Management Concept

As stated earlier, rewards and advancement are affected by 1) changes in required performance and 2) evaluation of the extent to which accompanying standards of performance are achieved. At the higher levels, two distinct types of specialists will in time emerge from engineering Capabilities Centers as a result of the "mixed performance required of these centers. The first will be the top-notch engineering manager and the second will be the top-notch technical expert. The engineering manager will become an expert in running programs and projects in his own area. The technical specialist will be developed by the Capabilities Center and utilized appropriately by Program Management as well as by the Capabilities Center. The result is that an alternative path of advancement in Program Management (in addition to the path in the Capabilities Center) is now available to the competent engineering manager. In fact, the engineering manager can, as a manager, develop in two somewhat distinct directions: he can become a manager within the Capabilities Center or he can move into broader program management. In the first case he has supervisory responsibilities, yet remains closely involved in the technical milieu. This choice represents a clear compromise between the typical engineer's professional needs and organizational achievement. On the other hand, in choosing the Program Management route, the engineering manager broadens himself by gaining an overview of company operation. His management duties involve coordination of many functional areas in respect to his specific program: engineering, manufacturing, marketing, procurement, quality assurance, and so on. The man who wants to remain purely a technical specialist may pursue advancement through the Senior Scientist (or Scientific) hierarchy It is interesting to note that, while much attention has been given by R&D organizations to the need for a "dual hierarchy" of advancement for scientific personnel, the Matrix concept in the manner described above in effect provides three channels for advancement of technical personnel.

Where Program Management is established as an avenue of advancement for competent engineering managers, it should be equal to (but not greater than) that through engineering. This objective can be achieved through:

- Establishing sound financial incentives.
- Encouragement of the best engineering managers both within and outside the company into Program Management.
- Successful completion of programs; acquiring of new business. The best engineering managers will try to join a winner.
- Intensive selection of program managers and associate program managers based on past project management success and on managerial skills.
- Carrying out development programs specifically for the enhancement of engineering-management abilities.

Not the least reward for the program manager is the satisfaction of "making things happen." He achieves his goals through other people and faces the challenge of coordinating and integrating the efforts of people with widely varying skills and interests. His satisfactions derive from attainment of results which are relatively observable: meeting time schedules, completing work within established budget, attaining customer approvals through meeting specifications, and so forth.

For the technical expert, the "professional ladder" becomes the basis of reward and advancement. A Senior Scientist hierarchy of advancement can be established as one offering prestige and status, as well as economic reward. The professional employee, the technical specialist, is more likely to be inclined toward outside professional identifications and interests. These interests should not be considered incompatible with company goals. On the contrary, the Program Management concept, with its "Capabilities Centers" and emphasis on technical excellence in the engineering groups, requires that the technical specialist do all in his power to develop his technical capabilities. This development depends directly upon maintaining lines of interest and communication with outside specialist groups and the scientific community as a whole. There is some evidence that management does not fully recognize the extent of professional commitment of the technical specialist. Survey results tend to show that scientists and engineers are somewhat less satisfied with their jobs than are those who have moved up the administrative ladder. It is recognized that engineers for the most part do not perceive the technical area as their best, long-range bet. More money, recognition, opportunity to influence decisions and achieve power are available through management. To offset this tendency, many electronics companies are investigating means of motivating and rewarding scientists and engineers through their technical hierarchy of advancement. The establishment of an equally rewarding ladder of advancement within the Capabilities Center for the technical expert provides an excellent means for overcoming these motivational difficulties which, in turn, will help to ensure the success of a Program Management operation.

Following these observations, efforts to develop better technical engineers should include the following:

- Development of a clear path of entry into and good advancement potential within the scientific hierarchy for qualified engineers and scientists.

- Greater encouragement of and recognition for technical achievements (e.g., patents, technical papers, state-of-the-art design, etc.).

- Communication downward of the broad objectives of the organization including its aim toward furthering technical leadership.

- Strong training and development programs aimed towards improving technical excellence. These include tuition reimbursement, in-house seminars, company-sponsored graduate fellowships, and similar programs.

- Understanding and acceptance of revised standards for performance of engineers--standards placing emphasis on technical excellence.

The latter action area is perhaps most important, particularly in the context of this article. How can the revised performance standards best be implemented in the organization? The answer depends in part on existing organizational procedures and programs in the company. For example, performance reviews, if carried out on a periodic basis, should be based on standards such as those outlined above for senior engineering people as well as program managers. Another means for implementation of these performance standards is in formulation and statement of program plans, such as those prepared by Program Management. Also, such requirement should be spelled out in revised job descriptions, as appropriate. In the latter case, it is particularly important that the revised job descriptions be made available to the individuals involved to help obtain their understanding and acceptance of job requirements.

Finally, a major area of implementation of performance standards is in a more or less formal goal setting program for higher level technical and managerial personnel. In such programs, which are being adopted more and more by industrial firms, individual job objectives are spelled out in writing for each key employee with yardsticks as standards for evaluation. Such a program provides an excellent means, first, for establishing an understanding of performance requirements and standards, and as a source of motivation for carrying them out as well. This is particularly true if the evaluation of results is linked in one way or another to a special financial reward such as a bonus or profit-sharing arrangement.

Most of the difficulties which are likely to arise from the changeover to a Matrix form of organization, with Program Management, result from misunderstanding concerning job responsibilities and standards for performance. Early recognition of the need to define new performance expectations, specific areas of responsibility (including overlapping or cooperative requirements), fair appraisal, and consideration of tailor-made reward and advancement systems will go a long way in facilitating engineering support in a change to the matrix organization.

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SYSTEMATIC ATMOSPHERIC REFRACTION ERRORS OF BASELINE-TYPE RADIO TRACKING SYSTEMS AND METHODS FOR THEIR CORRECTION

by

G.D. Thayer and B. R. Bean

ABSTRACT

The theory of systematic atmospheric radio refraction errors affecting measurements of range and range differences (and associated time rate of change of these quantities) is developed. It is shown that the refraction errors, particularly in range difference measurements, can seriously affect the accuracy of baseline-type tracking systems. A method is derived by which the systematic portion of these errors can be removed by means of linear relationships involving the surface value of the radio refractive index; the correction process devised can be used in real time if desired. Several test cases are examined where horizontally-varying profiles of the refractive index variation with height are used to calculate the errors, and the correction process based on surface refractive index values is found to be useful under these more general conditions. Approximately 98 percent of the total range or range difference error can be removed using this correction procedure. The problem of baseline optimization for deep-space tracking is examined briefly, and it is shown that a baseline length of about 4,000 miles is optimal for targets more than about 6,000 miles from the earth, and for such a system residual atmospheric refraction errors would be only a few hundredths of a microradian, assuming the validity of ray optics and of the models of the atmosphere used in this paper.

SYSTEMATIC ATMOSPHERIC REFRACTION ERRORS OF BASELINE-TYPE RADIO TRACKING SYSTEMS AND METHODS FOR THEIR CORRECTION

by

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1. Introduction and Background

Errors in measurements of distance made with radio equipment are caused in part by the refractive nature of the atmosphere: the velocity of propagation of a radio wave is a function of the refractive index structure over the propagation path, and the path itself is distorted from a straight line by refraction of the radio wave front. For the purpose of this paper a radio range error will be defined as the difference between the true distance separating two points and the distance as measured by the transit time of radio signals between the two points. In this definition it is assumed that the measured value of range is computed as if the radio signals were transmitted in a straight line and using the in vacuo velocity of light.

Radio range errors are of considerable importance in determining the overall accuracy of modern radio tracking or guidance systems which utilize the principle of triangulation (in three-dimensional space) to locate target position. The basic geometry involved in such triangulations is shown in figure 1. Two antennas, A_1 and A_2, each measure a range, R_1 and R_2 respectively, to a common target point. These two ranges, plus the (surveyed) baseline, B form the three sides of a triangle R_1BR_2 ; the direction cosine to the target (from A_1 with respect to B) can be easily derived from the trigonometric law of cosines as:

$$\cos\beta = \frac{R_1^2 - R_2^2 + B^2}{2R_1B} . \tag{1}$$

Phase-locked systems measure the range R_1 and a range difference,

$$DR \equiv R_1 - R_2 , \tag{2}$$

and in this case (1) may be rewritten in the form:

$$\cos\beta = \frac{DR}{B} + \frac{B^2 - DR^2}{2R_1B} . \tag{3}$$

The target position is determined by measuring another direction cosine with respect to a second baseline from A_1 to a third antenna A_3 (not shown).

The error due to refraction effects in the range difference measurement is simply the difference between the radio range errors of R_1 and R_2 in (2). Throughout the remainder of this paper the terms range error and range difference error will refer to atmospheric (not ionospheric) refraction errors which will be denoted by the operator Δ, while a range or range difference measurement containing a refraction error will be starred; thus from (2):

$$DR^* = R_1^* - R_2^* \quad ,$$

$$DR + \Delta DR = R_1 + \Delta R_1 - (R_2 + \Delta R_2) \quad ,$$

$$\Delta DR = \Delta R_1 - \Delta R_2 \quad . \tag{4}$$

The error in the direction cosine due to the range errors ΔR_1 and ΔR_2 can be found by deriving the differential of the function $\cos\beta$ in (1), since the range errors are always very small compared to the ranges ($\Delta R/R < 0.0004$):

$$\Delta(\cos\beta) \doteq \frac{R_2}{R_1}\ \frac{\Delta DR}{B} - \frac{B^2 - DR^2}{2R_1^2 B}\ \Delta R_1 \quad . \tag{5}$$

Equation (5) is asymptotically exact as the ratio $\Delta R_1/R_1$ approaches zero; (5) also reveals that there is no fundamental difference between the behavior of the phase-locked system and the independent range-finder system in that the largest portion of the direction cosine error in either case is a function of the range difference error. This is especially true for the case when $R_1 >> B$, since the term involving ΔR_1 in (5) may be ignored (note that $DR \leqq B$ must always be true).

Direction cosine information can be corrected for the refraction errors if the range errors, and in particular the range difference errors, can be corrected. The actual value of the radio range error occurring over a given propagation path is a function of the structure of the atmospheric radio refractive index, n, along and in the immediate vicinity of the path at the time that the range measurement is made. This refractive index structure is always to some extent an unknown quantity, but the most pronounced feature is always a quasi-exponential decrease of the refractivity, $N(n-1) \times 10^6$, with height above the surface.[1] It follows that radio range errors will show some systematic behavior depending, for example, on how much (if any) of the propagation path lies in the upper atmosphere where the N values are lower than near the surface.

In the sections that follow, the ray-geometrical equations for single path range errors are developed, and it is shown that systematic correction are possible based on the surface value of the refractivity, N_s. The systematic refraction errors of baseline triangulation systems, defined as those errors occurring in a horizontally homogeneous atmosphere, are derived, and it is shown how they may be corrected. The resulting correction technique is applied to some test cases involving horizontal inhomogeneity of the refractive index, and the results are in good agreement with the systematic predictions. Brief evidence is introduced that the systematic correction technique is also statistically optimum for correction of the non-systematic, random errors occurring under actual conditions. The results are extended to the case of very large baselines for tracking deep-space targets, and it is found that baselines on the order of 6700 km (4150 miles) are optimal (in the sense of minimal residual refraction errors) for tracking targets at distances of 10,000 km (6,000 miles) or more; the resulting rms residual direction angle errors range from about 0.01 to 0.05 microradians, assuming the validity of ray optics and of the models of the atmosphere used in this paper.

2. Theory and Correction of Systematic Radio Range and Range Difference Errors

The theory of radio range errors is most easily developed within the framework of ray-geometrical optics[2] (Fresnel approximation). In this notation the apparent radio range between two points is expressed as a line integral along the ray path connecting the points:

$$R_e = \int_0^{R_g} n\{s\}\, ds, \tag{6}$$

where n{s} is the refractive index as a function of position on the ray path, ds is the differential of ray path length, and R_g is the true length of the (curved) ray path between the points. Normally the origin of the ray path is taken to be the radio antenna and the upper terminal is the target position. Figure 2 shows ray tracing geometry. It is convenient to rewrite (6) as:

$$R_e = \int_0^{R_g} ds + 10^{-6} \int_0^{R_g} N\{s\}\, ds, \tag{7}$$

where N is the refractivity, $(n-1) \times 10^6$ in parts-per-million (ppm). The range error is now obtained by subtracting the true range, R_o, from the apparent range given by (7):

$$\Delta R = R_g - R_o + 10^{-6} \int_0^{R_g} N\{s\}\, ds. \tag{8}$$

The quantity $R_g - R_o$ in (8) can be called the geometric radio range error, ΔR_g, as it is caused only by the distortion of the ray path from a straight line, while the integral in (8) represents the error caused by the reduction in velocity of the wavefront in the refractive medium, and can be called the N-related range error, ΔR_N. Table I shows ray-traced values of both ΔR_g and ΔR_N for a typical N-profile ($N_s = 320$) and an extreme N-profile ($N_s = 400$). It can be seen that ΔR_N is the dominant quantity even at very small elevation angles.

Table I

Typical and Extreme Values of Range Errors for Targets Beyond the Atmosphere

	Typical $N_s \cong 320$			Extreme $N_s \cong 400$			Maximum %
θ_o Milli-radians	ΔR_g	ΔR_N	ΔR_e	ΔR_g	ΔR_N	ΔR_e	$\Delta R_g / \Delta R_e$
	Meters			Meters			
0	10	100	110	60	165	225	~27%
20	2.5	62.5	65	4.5	73	77.5	6%
50	0.7	38.1	38.8	1.0	43	44	2.3%
100	0.14	22.26	22.4	0.2	24.8	25	0.8%
200	0.02	11.9	11.9	0.03	13.0	13.0	0.23%
500	0.001	5.01	5.01	5.01	0.002	5.50	0.04%

A convenient quantity to study is the relative radio range error, defined as $\Delta R/R_o$, which is given in ppm by:

$$\frac{\Delta R}{R_o}\,(\mathrm{ppm}) = \frac{1}{R_o}\int_o^{R_g} N\{s\}\;ds + \frac{\Delta R_g \times 10^6}{R_o}\;, \tag{9}$$

or

$$\frac{\Delta R}{R_o}\,(\mathrm{ppm}) \cong \frac{1}{R_g}\int_o^{R_g} N\{s\}\;ds + \frac{\Delta R_g \times 10^6}{R_o}\;, \tag{10}$$

since R_g will not differ by more than about 10 ppm from R_o under non-ducting conditions.[8] As can be seen from (10), the bulk of the radio range error is essentially the average of $N\{s\}$ taken over the ray path.

The refractivity, N, may be written as the sum of two variables: the "expected value", $<N>$, which is essentially only a function of altitude, and a zero-mean random component N', whose statistical distribution may also be a function of altitude. Rewriting (10) with N defined in this manner, and the differential ds replaced by the equivalent dh csc θ (where θ is the local ray elevation angle and dh is the differential of height):

$$\frac{\Delta R}{R_o}\,(\mathrm{ppm}) \cong \frac{1}{R_g}\int_o^{R_g} <N\{h\}>\csc\theta\;dh + \frac{1}{R_g}\int_o^{R_g} N'\{s\}\;ds + \frac{\Delta R_g \times 10^6}{R_o} \tag{11}$$

The variable $<N\{h\}>$ is identified with very long time averages of N measured at a particular height h, above the surface in the vicinity of the ray-path origin (since $<N\{h\}>$ does vary to some extent with latitude, etc.). $N'\{s\}$ then represents the deviations of $N\{s\}$ from the expected values $<N\{s\}>$ which are taken to be the same as $<N\{h\}>$, where h corresponds to the height of the point s on the ray path. The $<N\{h\}>$ profile thus constitutes a sort of standard atmosphere: it has been found that standard atmospheres which are a function of the observed surface N value, N_s, are good predictors of refraction effects.[1, 3] Thus the "expected value" of range error, or what might be called the "systematic" range error, may be defined as the long time average of (11) for all cases where the N_s value is the same:

$$\left\langle\frac{\Delta R}{R_o}\right\rangle(\mathrm{ppm}) \cong \left\langle\frac{1}{R_g}\right\rangle\int_o^{R_g}\left\langle N\{h\}\right\rangle<\csc\theta>dh + \left\langle\frac{\Delta R_g}{R_o}\right\rangle\times 10^6,\; N_s = \text{const.}\;, \tag{12}$$

the expected value of the second integral in (11) vanishing since the $N'\{s\}$ have zero mean (the rms or second moment would not vanish, and defines the limit of accuracy of predicting $\frac{\Delta R}{R_o}$ from $\left\langle\frac{\Delta R}{R_o}\right\rangle$).

Applying the second theorem of the mean for integrals to (12) one obtains:

$$\left\langle \frac{\Delta R}{R_o} \right\rangle (\text{ppm}) \cong \frac{\langle \csc \theta_1 \rangle}{\langle R_g \rangle} \int_0^{\langle R_g \rangle} \langle N\{h\} \rangle \, dh + \left\langle \frac{\Delta R_g}{R_o} \right\rangle \times 10^6, \quad N_s = \text{const} \tag{13}$$

where $\langle \csc \theta_1 \rangle$ is $\langle \csc \theta \rangle$ evaluated at some point on the (mean) ray path. It turns out that for initial angles, θ_o, larger than about 10°, the value $\langle \csc \theta_1 \rangle$ is essentially just $\csc \theta_o$ (i.e. $\csc \theta$ is nearly constant over the part of the ray path where $\langle N\{h\} \rangle$ is much larger than zero). Thus the systematic range error is expected to be primarily a function of the integral of $\langle N\{h\} \rangle$ with respect to height. It is not surprising that this integral is a function of the $\langle N_s \rangle$ value, since that value defines the lower limit of the integral.

Figures 3 and 4 show ray-traced range errors for the CRPL Standard Radio Refractive Index Profile Sample*.[3] Figure 3 for $\theta = \pi/2$ radians (90°) shows the behavior of the integral of $N\{h\}$ with respect to height for each profile, while figure 4 shows the total range error at $\theta_o = 50$ mr (about 3°), both as a function of N_s. The regression lines show the best (linear) estimate of $\langle \Delta R \rangle$ as a least squares fit; the scatter of the individual points shows the effect of the integral of $N'\{s\}\, ds$ as in (11), plus whatever small error is incurred by predicting ΔR_g as a linear function of N_s for the $\theta_o = 50$ mr case. The reader should especially note the strong similarity of the distribution of points in figures 3 and 4; this indicates that the behavior of range errors at relatively small elevation angles is still primarily a function of the average of $N\{h\}$ over the target height interval, and that errors in predicting ΔR_g are of negligible effect at least for $\theta_o \geqq 50$ mr.

It will be noted that it was necessary to introduce a correction for station elevation above mean sea level because of the wide range of elevations encountered in the standard sample (8 m to 1908 m); for a single station this term would of course drop out, so it may be regarded as an adjustment to the constant term of the linear equation in N_s. The resulting correction procedure is a system of linear equation coefficients, such that:

$$\langle \Delta R \rangle = a\{h_t, R, h_s\} + b\{h_t, R\} N_s, \tag{14}$$

where h_t is target height, R is apparent range, and h_s is surface elevation above mean sea level.

The systematic refraction error of the baseline tracking system is given by (5) using expected values for ΔR, and ΔDR, the latter being defined as:

$$\langle \Delta DR \rangle = \langle \Delta R_1 \rangle - \langle \Delta R_2 \rangle. \tag{15}$$

The systematic errors are corrected for by use of the system of linear equations used to predict $\langle \Delta R \rangle$ as a function of observed N_s values. The residual error involved in correcting actual (real time) values of $\Delta(\cos \beta)$ by removing the systematic component $< \Delta(\cos\beta) >$ are of interest since they define the rms residual error in $\cos\beta^*$ after corrections have been made for observed N_s values. The residual variance of $\Delta(\cos\beta)$ produced by residual variances in ΔDR and ΔR_1 may be found by taking the statistical second moment of (5), leading to:

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* See Appendix

$$s^2\{\Delta(\cos\beta)\} \doteq \left(\frac{R_2}{R_1}\right)^2 \frac{s^2_{R_1} + s^2_{R_2} - 2 r_B s_{R_1} s_{R_2}}{B^2} + \left(\frac{B^2 - DR^2}{2R_1^2 B}\right) \times$$

$$\left[\frac{B^2 - DR^2}{2R_1^2 B} - \frac{2R_2}{R_1 B}\right] s^2_{R_1} + \frac{2R_2}{R_1 B} r_B s_{R_1} s_{R_2} \left(\frac{B^2 - DR^2}{2R_1^2 B}\right), \qquad (16)$$

where $s^2_{R_1}$, $s^2_{R_2}$ are the residual variances in predicting ΔR with $<\Delta R_1>$ and ΔR_2 with $<\Delta R_2>$, and r_B is the correlation coefficient between $\Delta R_1 - <\Delta R_1>$ and $\Delta R_2 - <\Delta R_2>$ for the particular baseline under consideration; the value of r_B is expected to be a decreasing function of increasing baseline length.

If it is assumed that the target is located at a range (R_1) very much larger than the baseline, B, then the last two terms of (16) may be neglected (and the last term of (5) as well), and R_2/R_1 approaches unity; in this case:

$$s^2\{\Delta(\cos\beta)\} \doteq \frac{s^2_{R_1} + s^2_{R_2} - 2 r_B s_{R_1} s_{R_2}}{B^2}, \quad R_1 \gg B. \qquad (17)$$

It is found both theoretically and experimentally that the residual variance of radio range errors after N_s correction tend to be proportional to the square of the expected value of the range error, $< \Delta R_1 >$, so (17) may be rewritten:[3, 4.]

$$s^2\{\Delta(\cos\beta)\} \cong \frac{s^2_{R_1}}{B^2}\left[1 + \left(\frac{<\Delta R_2>}{<\Delta R_1>}\right)^2 \; 2 r_B \left(\frac{<\Delta R_2>}{<\Delta R_1>}\right)\right], \quad R_1 \gg B. \qquad (18)$$

The systematic residual refraction error in the direction cosine can now be defined by the variance as given by (18), with the further assumption of horizontal homogeneity of the refractive index structure, which simply assumes $r_B = 1$ for all baselines:

$$s^2_H\{\Delta(\cos\beta)\} \cong \frac{s^2_{R_1}}{B^2}\left[1 + \left(\frac{<\Delta R_2>}{<\Delta R_1>}\right)^2 - 2\left(\frac{<\Delta R_2>}{<\Delta R_1>}\right)\right], \quad R_1 \gg B, \qquad (19)$$

where S_H^2 indicates the residual variance under the horizontal homogeneity assumption.

Numerical calculations have been performed to determine the systematic direction angle error of baseline systems under the stated conditions, using a modification of (5):

$$\Delta\beta = \frac{<\Delta R_1> - <\Delta R_2>}{B} \csc\beta\,, \quad R_1 \gg B\,. \qquad (20)$$

the results are shown in figure 5, together with total ray refraction which affects ordinary radar.[5] The residual standard deviations of the systematic errors were calculated from

$$S_H\{\Delta\beta\} \cong \sqrt{S_H^2\{\Delta(\cos\beta)\}\csc^2\beta}\,, \qquad R_1 \gg B\,,\ r_B \cong 1\,, \qquad (21)$$

where $S_H^2\{\Delta(\cos\beta)\}$ is given by (19). These errors are shown in figure 6.

Note that (19) yields a null result for cases when $<\Delta R_1> = <\Delta R_2>$, which occurs whenever the target is located equidistant from the two ends of the base line. In real systems there will be some residual error in all cases; this is because $r_B < 1$ for any real baseline. Thus it is logical to call the error contribution in (16) or (17) due to $r_B < 1$ the stochastic or random error of the system. Such errors can be derived theoretically, and are calculated in a forthcoming paper.[4] The results are presented here for comparison with figure 6, the total error having been calculated as:

$$S^2\{\Delta(\cos\beta)\} \cong S_H^2\{\Delta(\cos\beta)\} + \frac{S^2_{R_1}}{B^2}\left(\frac{<\Delta R_2>}{<\Delta R_1>}\right)2\left[1 - r_B\right]$$

and

$$S\{\Delta\beta\} \cong \sqrt{S^2\{\Delta(\cos\beta)\}\csc^2\beta}\,, \quad R_1 \gg B. \qquad (22)$$

These total residual errors are shown in figure 7. Also shown in figure 7 are total residual error estimates for tracking radar, ($R_B \approx 1$) where the random component (angular scintillation) has been found in two different ways:* by extrapolation of the baseline random component to zero baseline (the movable radar antenna sees essentially the direction cosine error since its direction angle is always 90°), and by empirical curve-fitting to the Collins Radio Company data.[5]

The accuracy of these systematic corrections when applied to actual (experimental) range error data has been examine in an earlier paper, with satisfacotry results.[3] For the shorter baselines, however, the random uncorrelated component of residual error is the dominant term in (22), and the errors shown in figure 7 are considerably larger than those shown in figure 6 for the baselines shorter than 300 km. It should be pointed out, however, that there is still a noticeable reduction (attributable to the N_s correction technique) in the overall rms errors for all baselines larger than about 3 km. In cases where the target is located relatively close to the baseline (i.e.: R_1 or $R_2 \approx B$), the N_s correction technique would be valuable even for very short baselines.

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*Shared baseline operation is discussed following (25).

3. Examination of Residual Direction Angle Errors for an Atmosphere Containing Horizontal Inhomogeneities

As a part of a study on the effects of atmospheric refraction on the precision of radio baseline triangulation systems, sometimes called "radio interferometers"[6], an analysis was made of the errors in range differences taken over rather long baselines in a horizontally inhomogeneous atmosphere. Four profiles were obtained, each from a number of radiosonde ascents made at approximately the same time at stations located roughly in a straight line covering a distance of about 750 miles. Figure 8 shows the two-dimensional profiles obtained in this way. Figure 8 is plotted in terms of A-units:[7]

$$A = N\{h\} + N_s \left[1-\exp\left\{\frac{h}{7}\right\} \right], \tag{23}$$

a convenient quantity which represents approximately potential surface refractivity, and thus emphasizes horizontal N-changes while masking the normal quasi-exponential vertical N-gradient. It is felt that these represent more the extremes of horizontal inhomogeneity than typical values, since the times for which the profiles for each path were assembled were chosen by a meteorologist using daily synoptic weather maps, attempting to find situations where the contrasts in air mass characteristics along the path were maximized.

Four hypothetical baselines were drawn on each of the four profiles, with baseline lengths of 3, 30, 300 and 1000 km, all with one common antenna location, and the second at the various distances specified. A hypothetical target was then assumed to be placed at an altitude of 70 km, and refraction analyses were performed with the target making a true elevation angle at the common antenna of 20, 50, 100, 200, and 500 milliradians (approximately 1°, 3°, 6°, 11½°, and 29°). The three kilometer baseline could not actually be analyzed as such because of limitations in the resolution which could be obtained from the graphical profiles, hence this baseline was analyzed from the common antenna ray path data as a horizontally homogeneous control case. Thus this horizontal inhomogeneity study applies only to the 30, 300, and 1000 km baselines.

Radio range errors were analyzed for each case by means of iterative ray-tracing techniques, and the range differences and resulting direction angle errors were calculated on a digital computer. In this connection, three important facts should be noted: 1) the direction angles were computed for the common antenna; 2) a standard refractivity atmosphere was assumed to exist at altitudes larger than those covered on the charts, as it was noted that horizontal changes did not seem to be significant above that point; 3) the target was assumed to be in a direction such that it would be located between the antennas of the longest baselines. The last was done to avoid putting the long baselines at too much of a disadvantage with respect to the short baselines, as the 70 km target height was already a disadvantage: at the 500 mr angle the secondary antenna of the 1000 km baseline has a look angle of only about 20 mr.

The results of these calculations are summarized in Table II. The rms direction angle errors for all four propagation paths are listed for each baseline and elevation angle, as well as the residual errors after correcting for the expected range errors with available N_s data. The residual errors are also shown as a percentage of the mean uncorrected direction angles, $\overline{\Delta\beta}$, listed in the table.

The N_s corrections were made using two methods, listed as I and II in the table. Method I consisted of making a purely systematic range difference correction by using the N_s value from the common antenna location to correct both range errors. Method II consisted of using separate N_s values at each end of the baseline* to correct the range errors independently: method II thus makes an attempt to correct for horizontal inhomogeneities between the two ray paths.

For comparison, theoretical values of residual direction angle error have been calculated using (16) and the second equation in (22). In these calculated from the tabulated values of $\overline{\Delta R_1}$ and $\overline{\Delta R_2}$, taking advantage of the empirically observed fact that for a target height of 70 km the rms residual error in predicting ΔR from N_s is 1.7% of the mean values of ΔR (for the CRPL Standard Sample).[3] The percentage errors were then calculated using the tabulated values of $\overline{\Delta\beta}$. These theoretical values were calculated for the extreme cases of $r_B = 0$ and $r_B = 1$ in (16); in the case of $r_B = 1$, or a perfectly homogeneous atmosphere, the residual percentage direction angle errors are all 1.7%, the same as for the individual range errors.

It is obvious from inspection that the residual errors for the shorter baselines are closer to the theoretical values for $r_B = 1$, and for the longer baselines closer to $r_B = 0$. The case of B = 1000 km, $\theta_o = 100$ mr, is close to the case mentioned previously where $<\Delta R_1> = <\Delta R_2>$, and is apparent that the N_s correction is not of great utility in this instance. Also of interest is that method I seems superior to method II for B = 30 km, and about even for B = 300 km. This implies that a single average N_s reading will be most useful in correcting errors for baselines of 30 km or less, using these methods.

It is possible to derive from the data going into this study the approximate observed values of r_B as a function of baseline length. Unfortunately, from such a limited sample the variations of observed values of r_B are quite large for each baseline, and not much confidence can be attached to even the mean for a given baseline. However, keeping this in mind, the value of r_B for the 30 km baseline seems to be in the neighborhood of 0.97 - 0.98 (which is probably too high), for the 300 km baseline about 0.4 to 0.6, and for the 1000 km baseline and about 0 to 0.4. These compare with theoretical values of 0.7, 0.3, and 0.13, respectively.[4]

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*As shown in figure 8.

Table II

RMS Range Difference Errors with Horizontally Inhomogeneous Refractivity Structure; for $h_T = 70$ km.

Baseline (km)	Elev. Angle (mr)	$\overline{\Delta R_1}$ (m)	$\overline{\Delta R_2}$ (m)	$\overline{\Delta\beta}$ (μrad)	$S(\Delta\beta)$ method I (μrad)	$S(\Delta\beta)$ method II (μrad)	$\%S(\Delta\beta)$ I	$\%S(\Delta\beta)$ II	Expected $\%S(\Delta\beta)$ $r_B \equiv 0$	Expected $\%S(\Delta\beta)$ $r_B \equiv 1$
3	20	56.1	55.6	12,655	179	--	0.5%	--	240%	1.7%
	50	36.2	35.9	2,084	38.5	--	3.3%	--	280%	"
	100	21.7	21.5	634.5	8.40	--	2.2%	--	270%	"
	200	11.7	11.5	213.9	3.83	--	1.7%	--	220%	"
	500	4.95	4.86	63.80	0.832	--	2.4%	--	130%	"
30	20	56.1	51.3	8,624	134	408	1.6%	4.7%	27%	1.7%
	50	36.2	33.2	1,823	91.2	116	5.0%	6.4%	28%	"
	100	21.7	19.8	561.8	20.0	29.9	3.6%	5.3%	27%	
	200	11.7	10.4	183.8	6.29	8.26	3.4%	4.5%	16%	
	500	4.95	4.06	50.75	1.34	1.42	2.6%	2.8%	12%	
300	20	56.1	23.7	1,457	25.6	44.7	1.8%	3.1%	3.2%	1.7%
	50	36.2	15.1	473.2	30.1	29.4	6.4%	6.2%	3.2%	
	100	21.7	7.6	130.2	1.42	3.14	.1%	2.4%	2.8%	
	200	11.7	2.4	32.1	0.462	0.634	1.4%	2.0%	2.2%	
	500	4.95	6.69	16.31	1.38	1.17	8.5%	7.2%	8.1%	
1000	20	56.1	6.8	489.6	5.91	8.15	1.2%	1.7%	1.9%	1.7%
	50	36.2	12.7	241.8	2.77	5.32	1.2%	2.2%	2.8%	
	100	21.7	22.2	8.86	8.74	4.52	98.6%	51.0%	91.5%	
	200	11.7	38.8	317.5	20.8	7.65	6.6%	2.4%	2.5%	
	500	4.95	66.9	803.3	48.0	19.1	6.0%	2.4%	1.8%	

The most important conclusion derived from this particular study is that for small baselines ($B > 30$ km, or perhaps more) it is important to use a single value of N_s to predict the range errors for all antennas in the system, regardless of the number of instruments available for measuring N_s. In cases where a number of accurate instruments are available, the best procedure would seem to be to use the mean of all readings at any given time. For very long baselines, such as are advocated in Section 4 of this paper, it is important to correct the range errors for each antenna independently since for such separations r_B approaches zero. The reader can no doubt draw many more interesting conclusions from a careful examination of the data in Table II, as regards such things as the relative advantages of varying baselines under different target position configurations, etc.

4. Optimum Baseline Systems for Deep Space Tracking Applications

If a tracking system were to be built with two of the antennas located at opposite points on the earth's surface, i.e., a surface baseline of about 20,000 km, the system would only be able to "see" targets at extremely large ranges located very close to a plane perpendicularly bisecting the baseline, which would be a diameter of the earth. Such a system obviously would not be practical, but even more important, the residual tracking errors of such a system would be larger than for systems of somewhat smaller baselines because both ray paths would always be near zero initial elevation angles, and hence would traverse a portion of the earth's atmosphere effectively some 50 to 70 times "thicker" than that penetrated by a ray at near-vertical incidence. The question then arises as to what might constitute an optimum baseline length in terms of minimum residual direction angle error as a function of target range and elevation angle.

A theoretical study has been made of residual tracking errors for a large number of target positions and baseline lengths using a digital computer. In order to obtain results which would be as unbiased as possible, the target positions were restricted to heights beyond the earth's refractive atmosphere, the the calculations were made using geocentric target angle and true target height as target-to-baseline geometric parameters, as shown in figure 9. As can be seen the geocentric target angle is essentially the zenith angle referred to a point at the center of the baseline, but measured at the center of the earth. A value of $\alpha_T = 0$ usually for any baseline at any given target height. The residual angle error for each case was determined as follows:

1. From the true geometry involved the true elevation angle (with respect to the earth's surface) was calculated for each antenna. The target is assumed to lie in a plane containing the baseline and the center of the earth.

2. The residual single-path range error for each antenna ray-path was then calculated using an empirical relationship

$$S_R = \frac{\pm 4.0 \times 10^{-5} (\mathrm{km})}{\sin\theta + \dfrac{2.94 \times 10^{-2}}{1 + 14\theta + 43\theta^2}} \qquad (24)$$

where θ is the true elevation angle at the antenna in radians. This equation is quite accurate (error less than 1%) in giving the residual range error as determined by linear regression of ΔR versus N_s for the CRPL Standard Radio Refractive Index Profile Sample[3]. Note that the residual error at vertical incidence is given as ±4.0 cm by (24).

3. The residual direction angle error was then calculated by solving (16) and (22) using (24) for the values of S_{R_1} and S_{R_2}, and using a theoretical form for r_B.[4] The function r_B is approximately zero for $B > 2500$ km.

Figure 10 shows some of the results of this study for target heights of 10^7 km to infinity. The curve marked "locus of critical points" represents the residual error for each baseline at the limiting geocentric target angle beyond which the target drops below the horizon for one of the antennas (the curve for a short baseline could, of course, cross this region if plotted). For $\alpha_T = 0$ the 10,000 km baseline is seen to be optimal, whereas for increasing values of α_T successively shorter baselines become optimal. It is obvious that baselines much longer than 10^4 km will never be optimal, since at lesser target ranges the shorter baseline is at even more of an advantage. In particular the impractical earth's-diameter system mentioned previously appears as a single point (labeled 20,000 km), and has a value of σ_ϵ ($S\{\Delta\beta\}$ in (22)) over 10 times as large as the more optimal baselines. It should be noted that the curve marked 6680 km is nearly optimal (within ± 14% of optimum σ_ϵ) over a rather large range of α_T, from zero to nearly $\pm 50^\circ$; the significance of this particular baseline length will be explained shortly.

Results for a true target height of 10,000 km are shown in figure 11. The limiting baseline is now seen to be about 15,000 km. The 6680 km baseline is seen to be optimal for this case up to geocentric target angles of about $\pm 30^\circ$. The general level of the curves in the vicinity of $\alpha_T = 0$ is not much higher than for the case of near-infinite target range, but the shape of the curves is much "tighter" with respect to α_T; this is because of the "foreshortening" effect of the relatively close target position (close with respect to the baselines involved).

The results of this study can also be presented as a function of baseline with constant h_T or α_T, and the other as a curve parameter. Such a plot is shown in figure 12 for a constant geocentric target angle of $\pm 23^\circ$. The locus of critical points represents the limiting value of baseline for each target height at $\alpha_T = 23^\circ$, and the cross-hatched zone now represents a truly "forbidden" region. The 6680 km baseline (dashed vertical line) is now seen to be optimal or nearly so for target heights ranging from about 10,000 km to infinity. The angle of 23° is chosen since it is nearly equal to the earth's equatorial inclination ($23^\circ 27'$).

A deep-space tracking system would presumably be used for tracking/ guidance of such things as lunar or interplanetary probes, (e.g., Mariner, Pioneer, etc.). This implies that the target of such systems would be generally located close to the plane of the ecliptic, which varies in declination (angle referred to earth's equatorial plane) by ± 23°27'. The moon and all but one of the planets are restricted to declinations of about ± 30° or less, the lone exception being Pluto with ± 40° limits [cf. American Ephemeris & Nautical Almanac, various years]. Hence it appears that the logical location for a deep space tracking system with continuous tracking capabilities for deep space probes would be a series of east-west baselines located along or near the equator, with north-south baselines straddling them. The north-south baselines would then be required to track normally within a geocentric target angle of only ± 30°. The required number of east-west baselines ranges from a minimum of two for the limiting case of zero baseline, each working at $\alpha_T = \pm 90^\circ$, to infinity for the other limiting case of 20,000 km baseline where each works only at $\alpha_T = 0$. The required number is given by the formula

$$j = \frac{2}{1 - \frac{B}{\pi r_o}} \tag{25}$$

where B is the baseline measured along the earth's surface and r_o is the radius of the earth; any fraction must be rounded off to the next larger whole number. Obviously the smallest practical value of j is three, for which the value of B/r may range up to $\pi/3$ or 60°. At this point an interesting thing happens: one in effect has six different baselines; each of the six antennas of the original three east-west baselines being shared with one of the intervening baselines. This effect is shown in figure 13. Such a system can logically be called a "shared baseline", and will be so referred to in the remainder of this paper. The particular system illustrated in figure 13 is called "optimum shared baseline" because of the generally small tracking errors over the necessary range of target positions, as has already been pointed out, and because it requires only the practical minimum in equipment. The surface baseline of such a system is about 6680 km or 4150 miles. A shared baseline configuration with eight baselines and eight east-west antennas can be used with the resulting 5000 km baselines used for the smaller target ranges, and double baselines of 10,000 km used for large target ranges, but the residual error advantages over the six-baseline system would be very slight and probably not worth the extra cost involved.

The optimum shared baseline system must track equatorially only to geocentric target angles of ± 30° before an adjacent baseline picks up the target, and has already been pointed out the north-south baselines will probably not be required to track at over ± 30°. The effect on either baseline of tracking a target off 30° both in declination (north-south geocentric angle) and in east-west geocentric angle is to increase the effective geocentric angle to about ± 35°. This allows continuous tracking with a 6680 km baseline from a target height of about 10,000 km, (6000 miles), since the limiting angle for this case is about ± 38° (see figure 11).

To illustrate the overall tracking capabilities of the optimum shared baseline configuration, figure 14 has been prepared showing total target position error for several different target ranges (or heights) as the target passes over three of the east-west baselines at two declination angles. The total position error was calculated as the root-square-sum of the residual range error and the north-south and east-west position error (calculated as direction angle error times range):

$$\sigma\{P\} = \sqrt{\sigma_r^2\{R\} + R^2[\sigma_\epsilon^2(n-s) + \sigma_\epsilon^2(e-w)]}$$

The switching effect from one baseline to the next is clearly shown, especially for the R = 10,000 km case. The shared baseline effect is also illustrated in figure 7.

An example of a system of this type in use at the present time is the NASA Deep Space Instrumentation Facility (DSIF), which consists of three range and range rate stations located about 8000 miles apart near Goldstone, California, Johannesburg, South Africa, and Woomera, Australia. These independent trackers are capable of providing elevation and azimuth information with a precision on the order of 300 microradians, however when tracking a deep space probe, such as Mariner II, which follows a smooth orbit, advantage can be taken of redundancy techniques to utilize the four shared baselines inherent in the system. In this manner the trajectory of Mariner II was corrected to within a 10 mile uncertainty at closest encounter with Venus, at a range (from earth) of about 36 million miles, a residual error of only 0.28 microradians.[8] Figure 14 shows that under these conditions the theoretical minimum refraction error of the optimum shared baseline system would yield a position error of only about ± ½ mile, however the DSIF system accuracy was limited by range resolution and survey errors rather than refraction effects.

5. Conclusions and Discussion of Results

In summary, it has been shown that:

1.) It is possible to correct systematic range and range difference errors with a system of linear equations based on the use of surface refractive index observations.

2.) Under conditions of marked horizontal inhomogeneity of the refractive index structure the systematic corrections are still very useful, with the residual range errors increasing only about 30% over those for horizontally homogeneous conditions.

3.) For baselines of less than about 30 km (20 miles), all system range errors should be corrected using a single, averaged, N_s reading (theoretical considerations based on turbulence theory indicates that N_s readings should be averaged over a period on the order of 1½ to 2 hours for best results--see Reference 4).

4.) The optimum baseline lengths for deep space tracking are on the order of 4000 miles; potential tracking accuracies are a few parts in 100 million of target range.

The construction of baseline systems on the order of thousands of miles of separation poses a number of technical problems, in particular synchronization problems for cw systems which are presently causing difficulties over much shorter distances.[9] The solution would seem to lie in the direction of a number of independent range-only trackers.[10] There is also the problem of geological stability of the baselines: a shift of only five inches in 4,000 miles of the earth's crust would cause errors comparable to the residual refraction errors of such systems. However, it is possible that the use of artificial satellites for both baseline-calibration and synchronization between trackers could solve most of the problems associated with such large baselines. As an example of this technique, during the flight of Mariner II Jet Propulsion Laboratory Personnel, utilizing redundancy techniques, were able to use DSIF tracking data to reduce survey uncertainties for the three stations from 100 yards to 20 yards.[8]

6. Acknowledgements

The authors wish to acknowledge the general assistance of the staff of the Radio-Meteorology group, and in particular of L. P. Riggs who prepared the meteorological data for section 3.

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Appendix

The CRPL Standard Radio Refractive Index Profile Sample

The CRPL Standard Sample consists of 77 refractive index profiles obtained from radiosonde data for 13 U. S. Weather Bureau stations. The 13 stations were chosen to represent a cross-section of the climatic types found in the U.S.A. and vicinity, and range from semi-tropical (Miami, Fla.) to sub-arctic (Fairbanks, Alaska); the stations range in elevation from near sea level to over 6,000 feet (Ely, Nev.). Six profiles were chosen from individual radiosonde flights for each station, each of the six representing a particular type of profile: 26 of the profiles represent rather average conditions, while the other 51 each contain one or more abnormalities (i.e.: surface duct, super-refractive ground layer, elevated duct, or a combination of these). This sample has been found to provide reliable estimates of refraction effects (as a function of N_s), as well as the variations and residual errors (after N_s correction) of these effects, when compared to actual data (see reference 3).

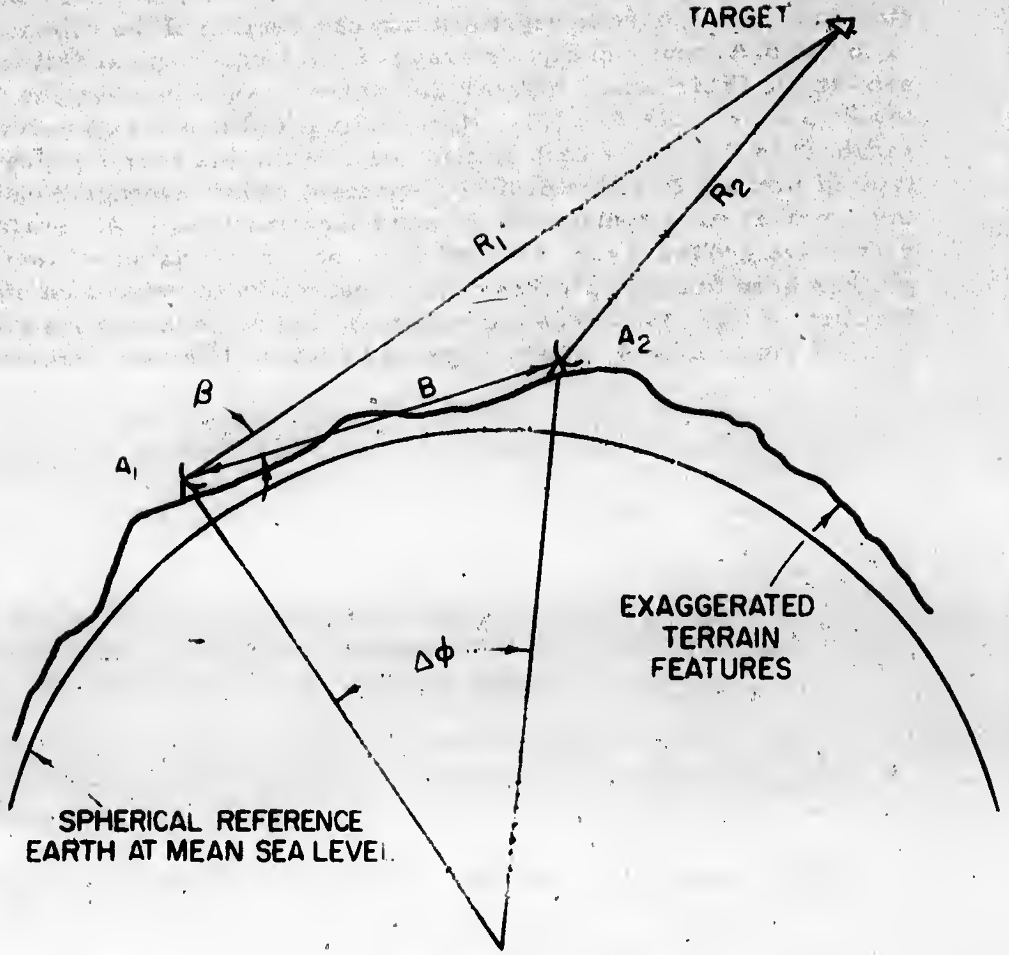


FIG 1 BASELINE TRACKING SYSTEM GEOMETRY

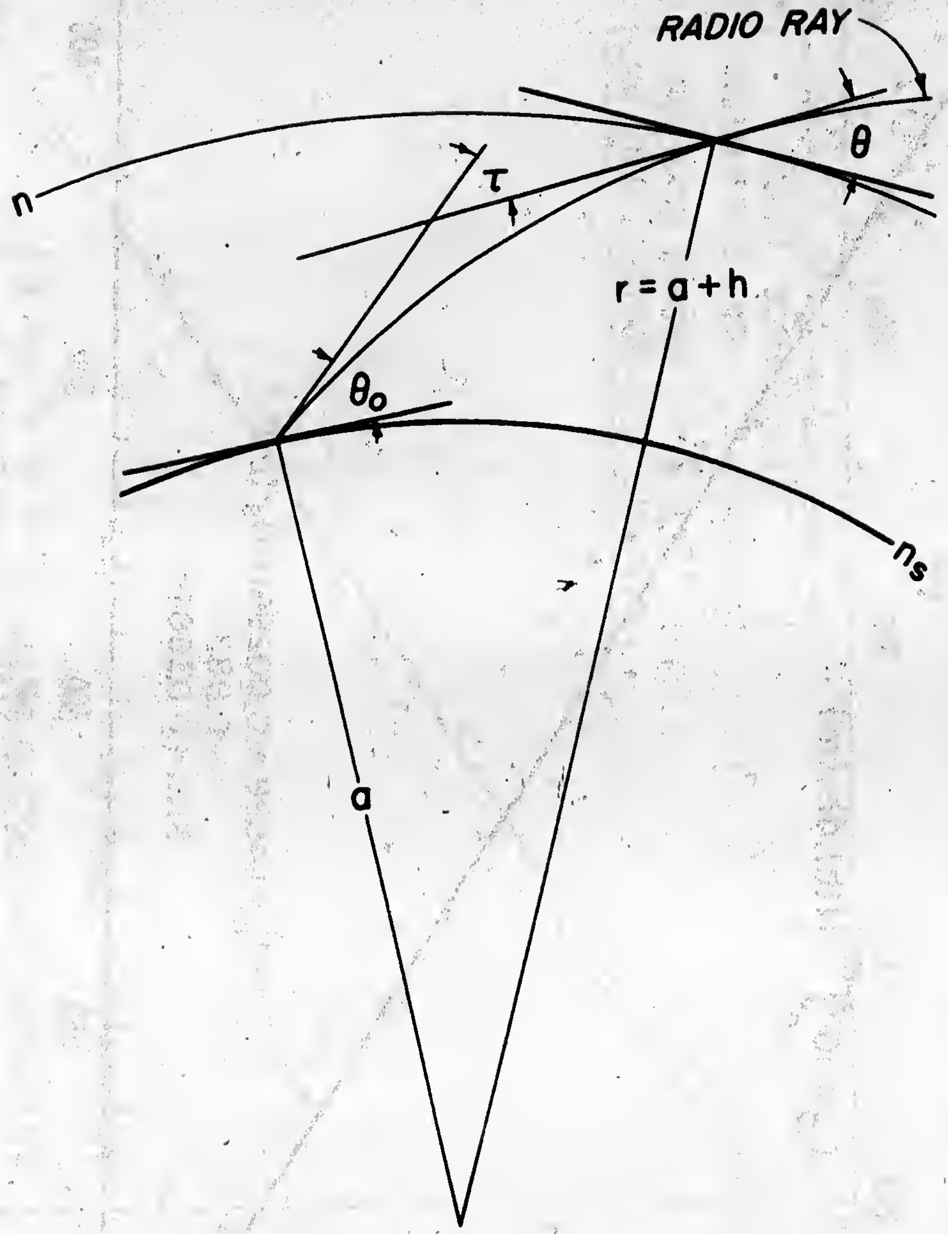


Figure 2. Geometry of radio ray refraction.

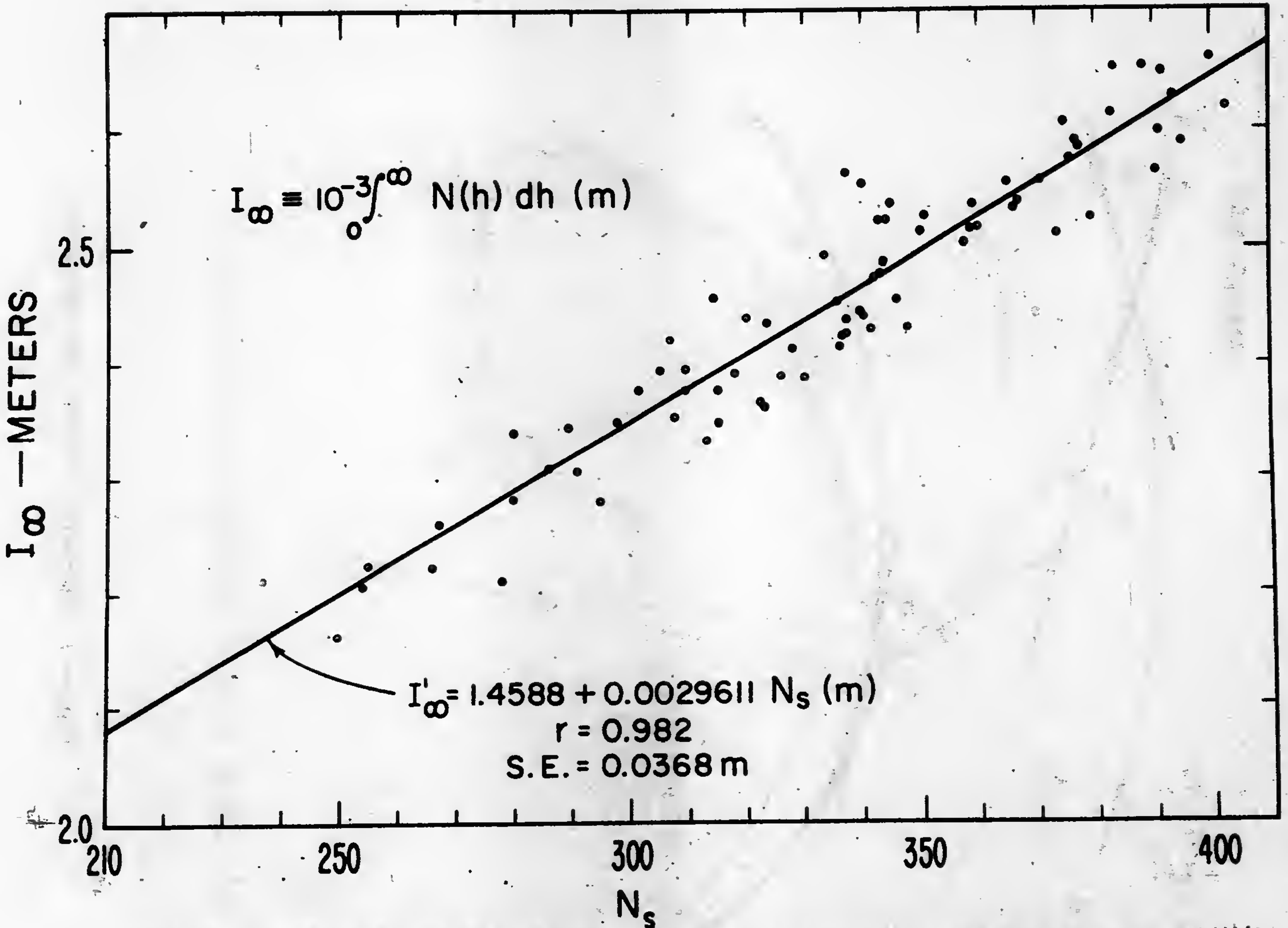


Figure 3. Integrated refractive index profiles for the CRPL Standard Sample. The integral of N(h) with respect to height is taken from the surface to 70 km, above which point N(h) ≡ 0. The line represents the least squares regression of I'(∞) on N_s, where I'(∞) is the value of the integral as adjusted for the dependence on station (surface) elevation above mean sea level

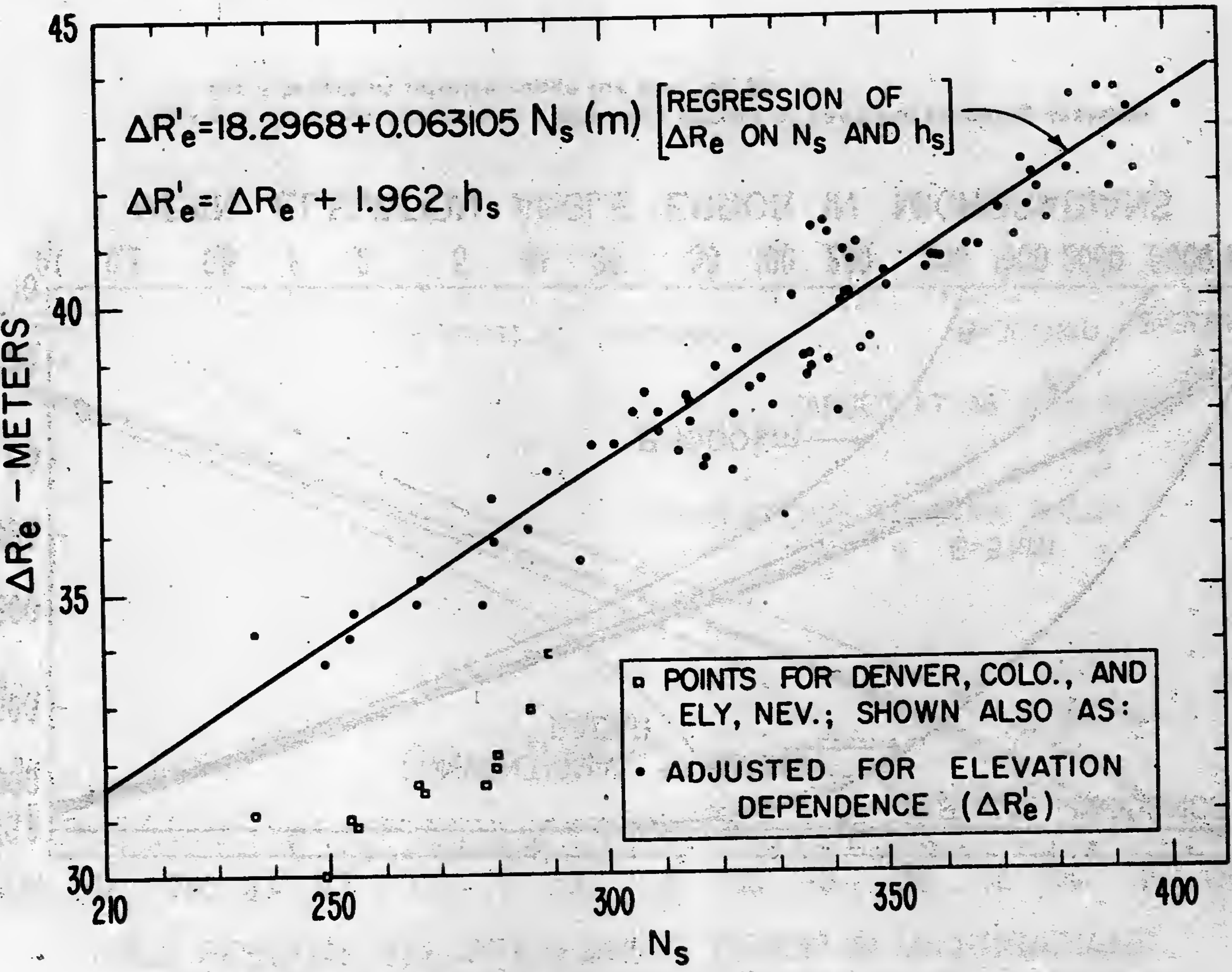


Figure 4. Total range error at $\theta_o = 50$ mr, $h_t = 70$ km, for the CRPL Standard Sample. The line indicates the least squares regression of $\Delta R'_e$ on N_s where $\Delta R'_e$ is the total range error adjusted for the dependence on station elevation.

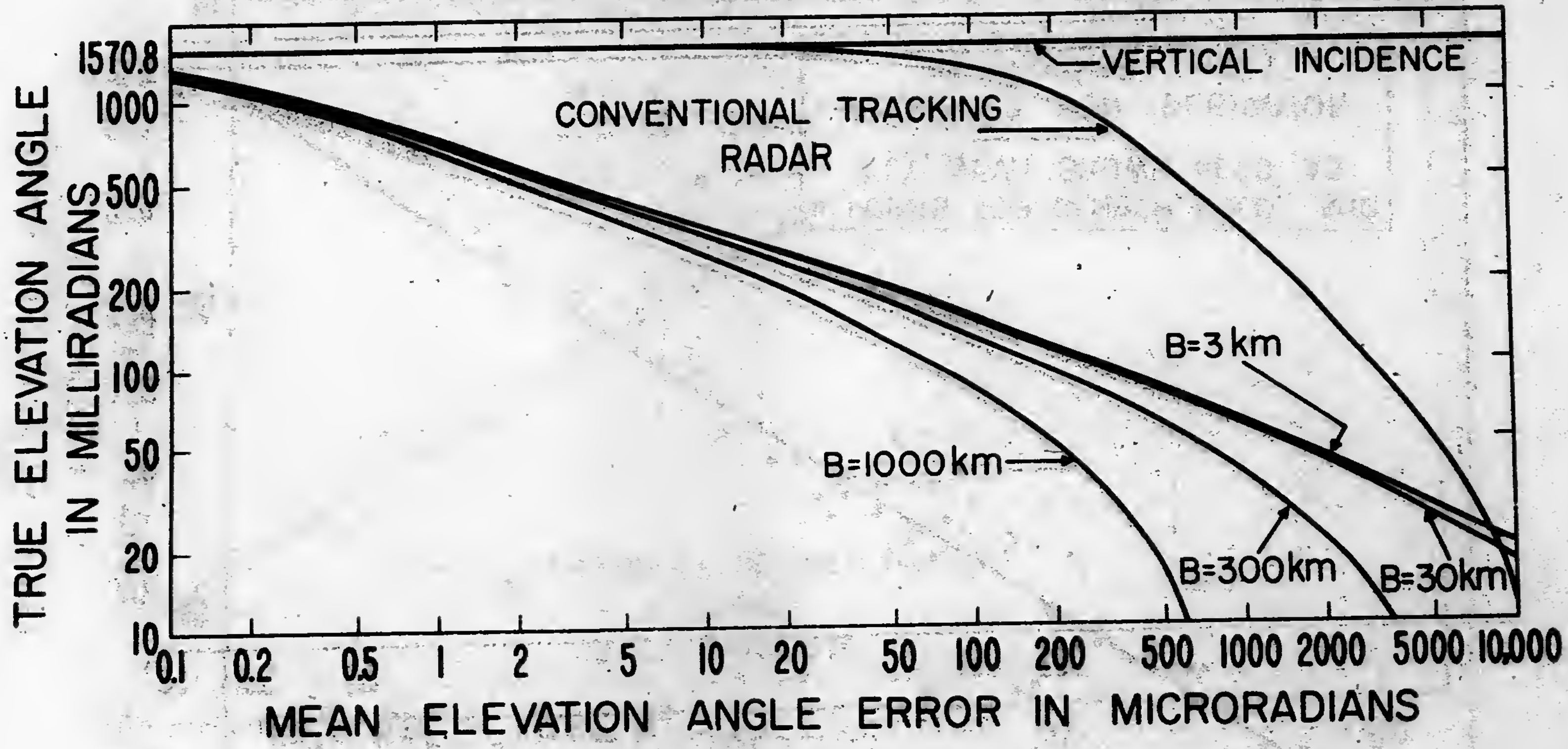


Figure 5. Average systematic refraction errors of baseline tracking systems for a target at infinite range (or R > 100 B).

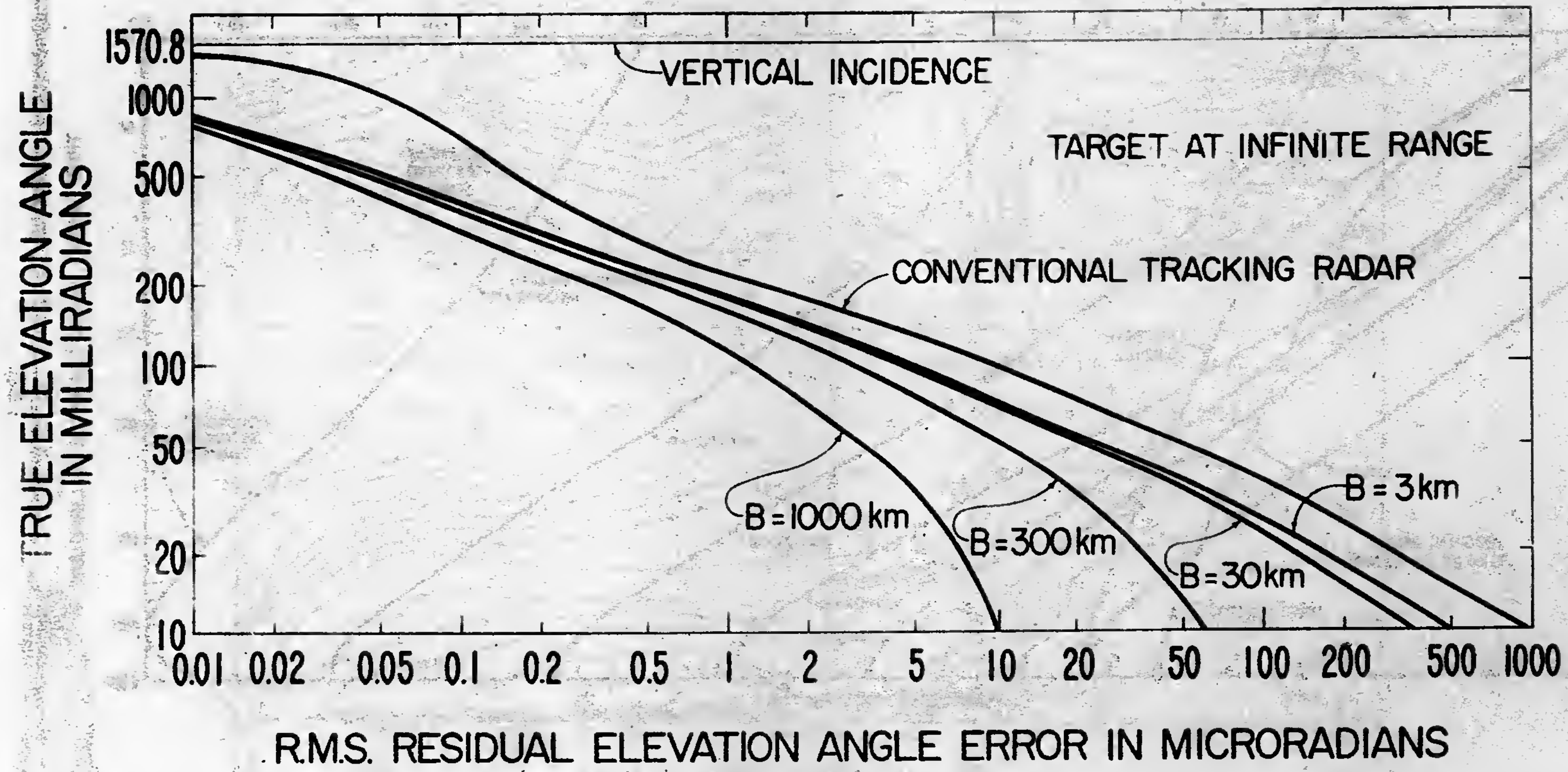


Figure 6. Residual rms systematic refraction errors of baseline tracking systems, target at infinite range.

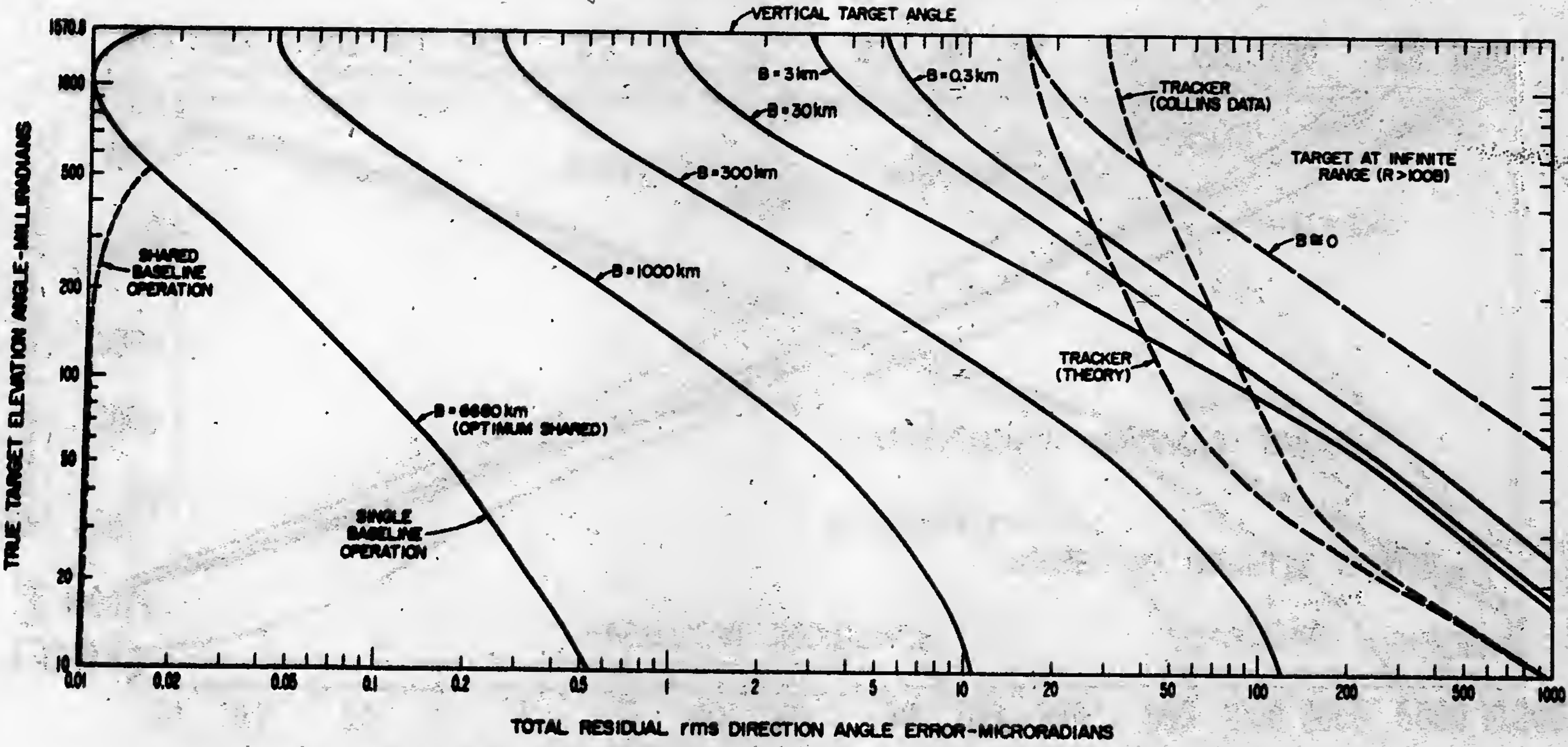


Figure 7. Total systematic and random residual baseline tracking system refraction errors, target at infinite range. Total of systematic and random (angular scintillation) errors of tracking radar also shown, both theoretical (extrapolated zero baseline data) and observed (from Collins Radio Co. data).

B-45016

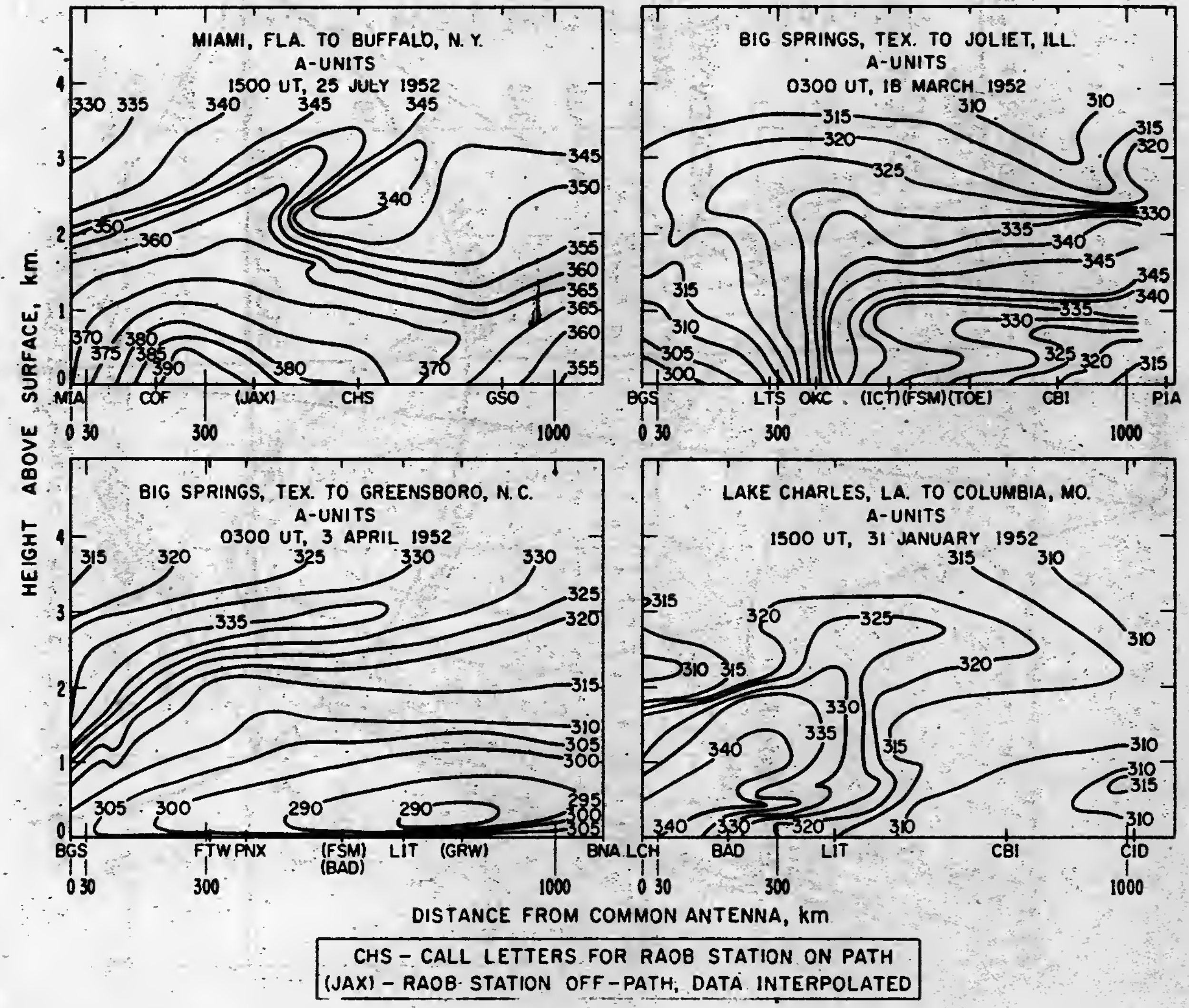


Figure 8. A-unit refractive index structure derived from radiosonde data along four hypothetical baseline paths.

B-63130

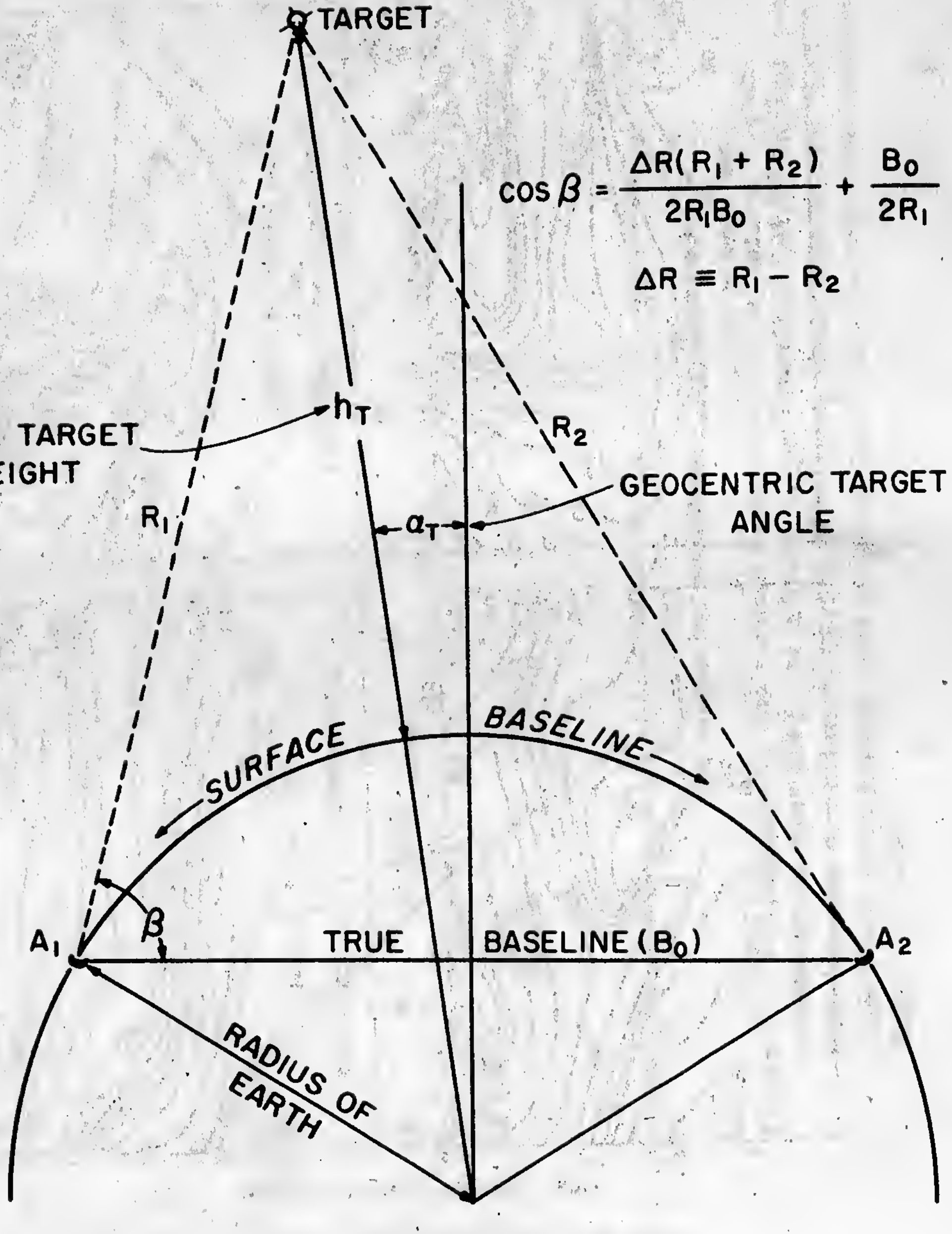


Figure 9. Deep-space baseline tracking system geometry, showing true target position parameters α_t and h_t.

B- 63419

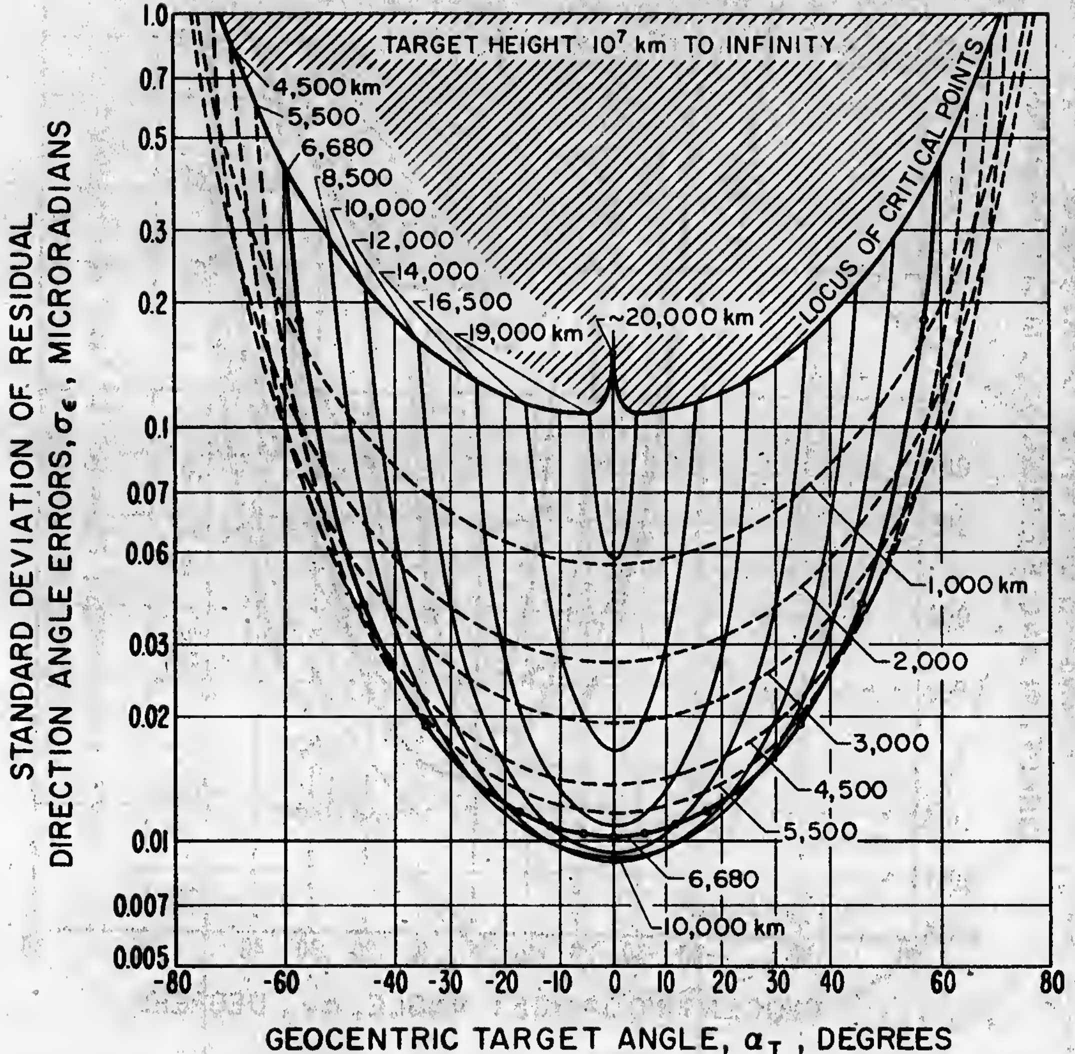


Figure 10. Residual direction angle errors of long-baseline tracking systems at very large target ranges.

B- 43425

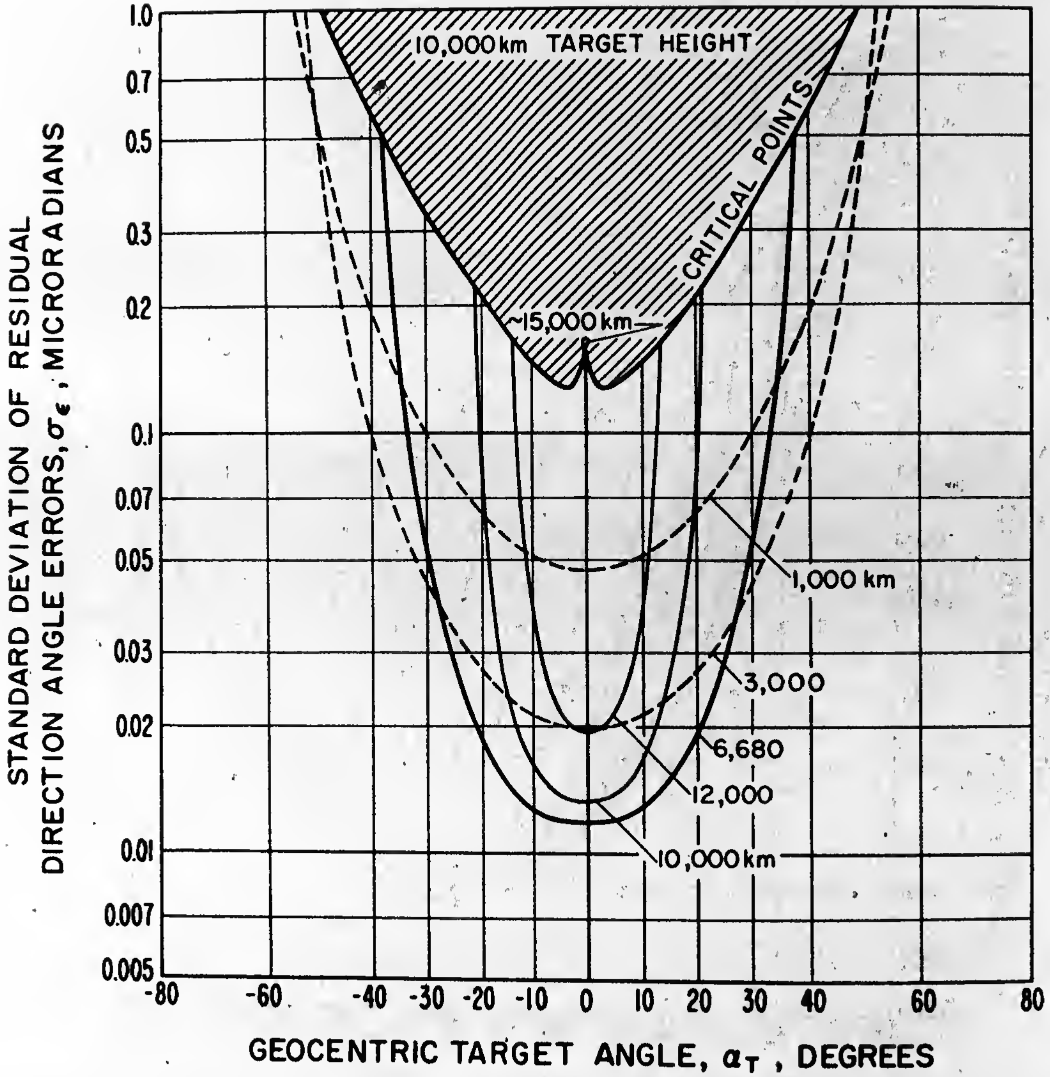


Figure 11. Residual direction angle errors of long-baseline tracking systems at a 10,000 km target height.

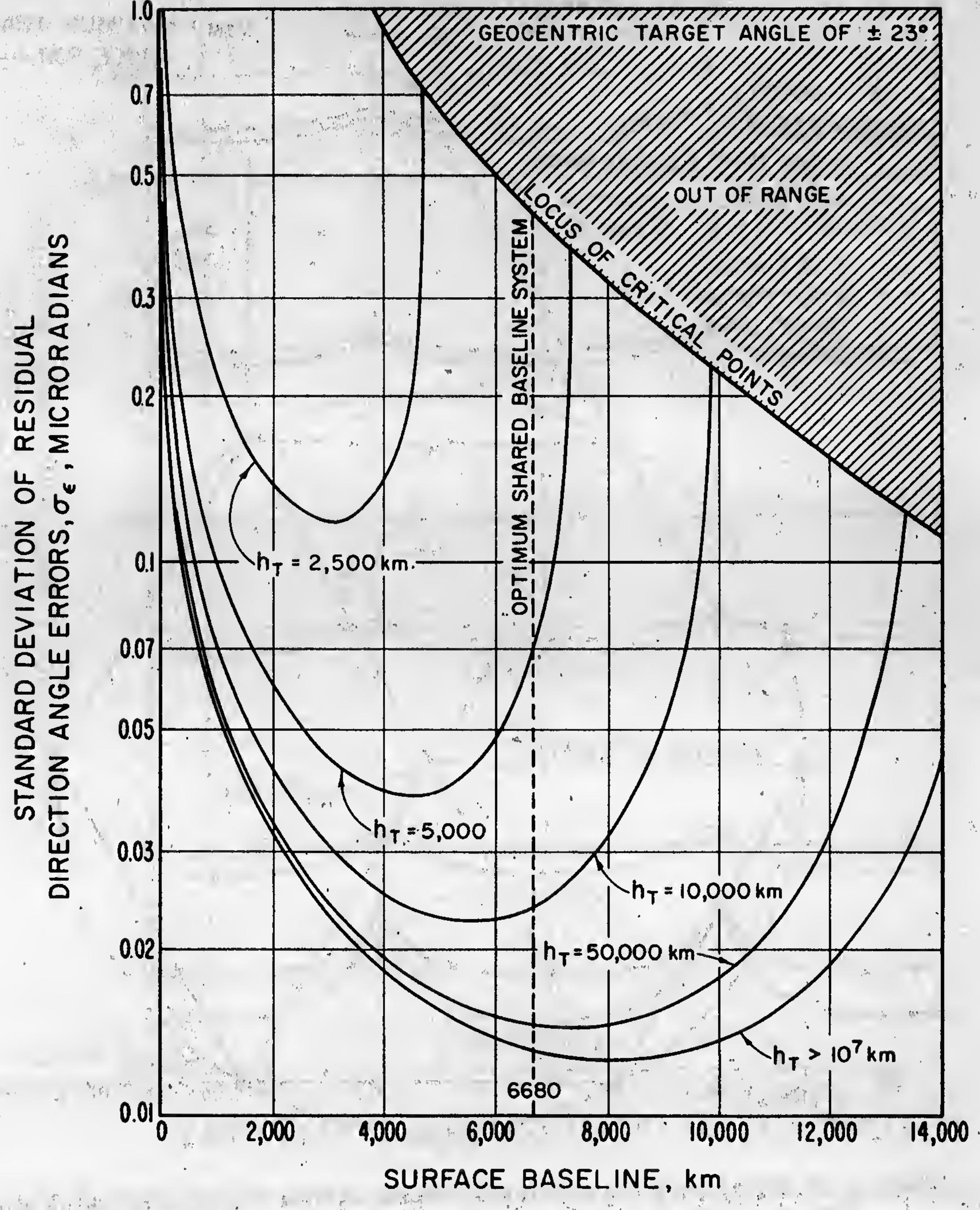
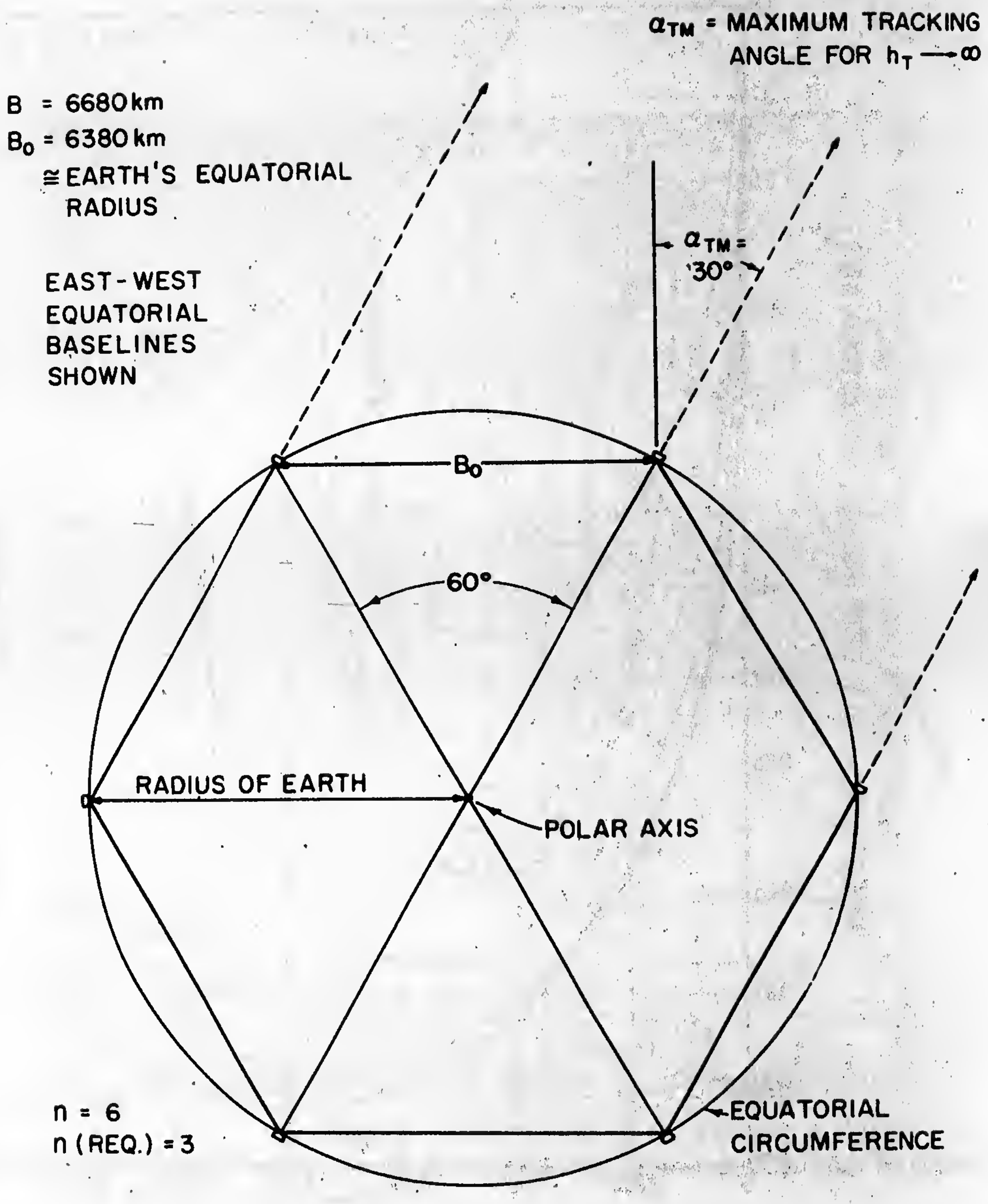


Figure 12. Residual direction angle errors of long-baseline tracking systems at a geocentric target angle of ± 23°.



α_{TM} = MAXIMUM TRACKING ANGLE FOR $h_T \rightarrow \infty$
B = 6680 km
B_0 = 6380 km ≅ EARTH'S EQUATORIAL RADIUS
EAST-WEST EQUATORIAL BASELINES SHOWN
α_{TM} = 30°
B_0
60°
RADIUS OF EARTH
POLAR AXIS
n = 6
n (REQ.) = 3
EQUATORIAL CIRCUMFERENCE

Figure 13. Polar plan view of the optimum shared baseline tracking system configuration, showing only the east-west equatorial baselines.

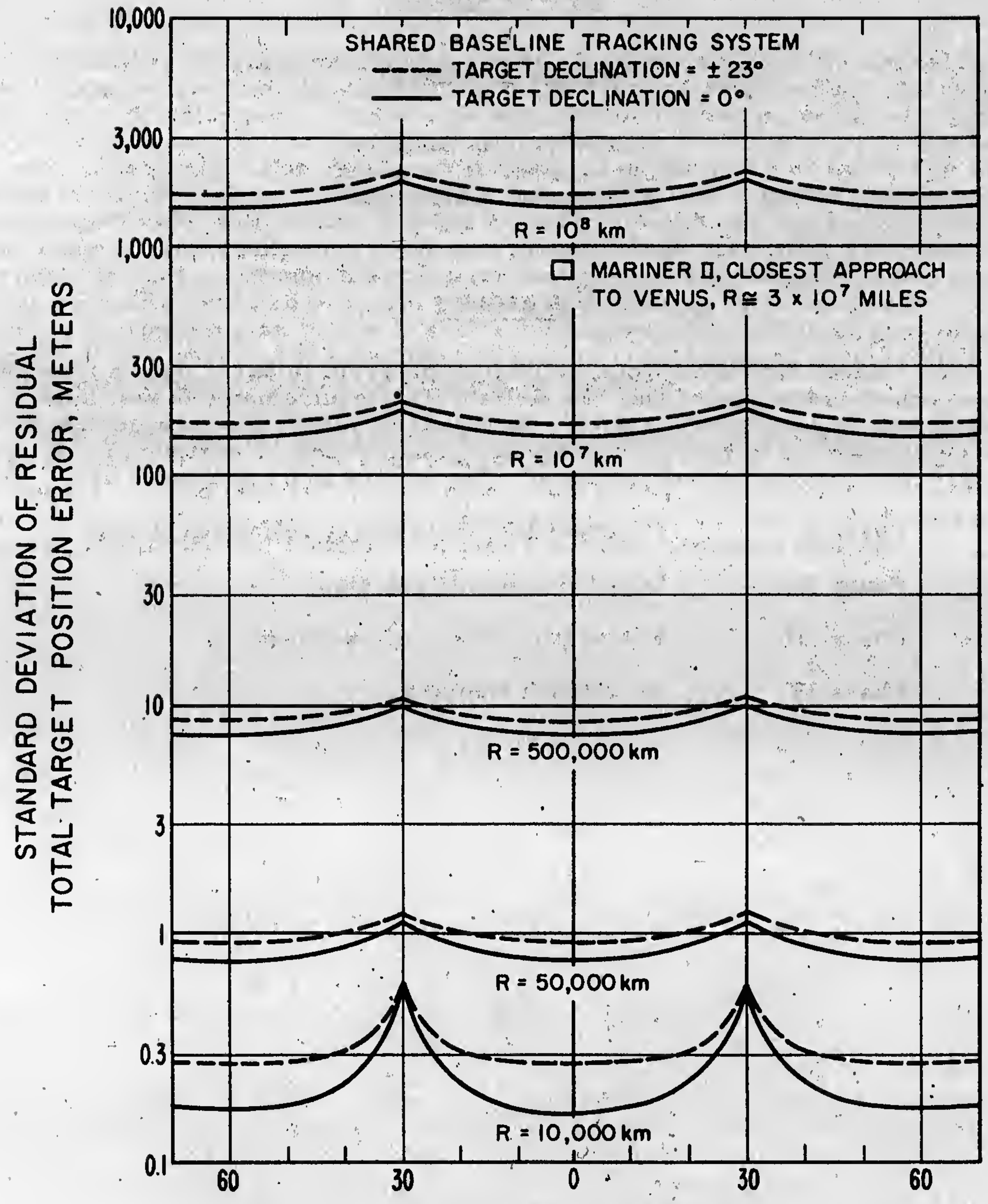


Figure 14. Residual rms total target position error for the 6680 km optimum shared baseline tracking system configuration.

A SPACECRAFT TOTAL RELIABILITY PROGRAM

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SUMMARY

The subject spacecraft total reliability program is based upon a time-phase sequence scheduled to insure that "Hi-Reliability" requirements of soft lander spacecraft can be practically applied. Such phasing allows for management planning to be responsive to required customer specifications or customizing of existing programming for customer approval. The phases to be discussed are:

Phase I	Proposal and preliminary reliability design
Phase II	Detailed reliability design
Phase III	Fabrication, test and operation
Phase IV	Postlaunch analysis

INTRODUCTION

Reliability program management to be effective requires the equivalent detailed management attention that is applied to Systems Engineering, Operations and Hardware Development. To provide such detailed attention requires a definite time-phased sequencing of events which interlock the reliability aspects into the total system program. For spacecraft (unmanned or manned and especially those required to survive surface impact, i. e., soft lander, and requiring extended space mission objectives), this means that the reliability objectives are necessarily high and the controls and disciplines applied to maintain high reliability very exacting. Experience in such environments indicates that while many tradeoffs may exist, they are normally expected program pertubations and will cause only minor changes in a program plan that is based on fundamental principles of good planning. The intent of this paper is to consider the unusual constraints of spacecraft in terms of sequencing and provision of program planning.

The means of estimating the effect of such phasing by management is through milestones achieved in each phase and ultimately in system hardware performance in simulated and actual mission operation. Postlaunch analysis and failure review techniques are primary feedback circuits for next launch corrective action and may represent a significant effort if major problems appear.

CONCEPT

Hi-reliability spacecraft objectives are closely bound to the successful completion of mission objectives which may include many severe and demanding constraints such as:

- Deep space environment
- Extremely high and low temperatures
- Radiation
- Meteorite collision
- Precision guidance tolerances
- Long mission times
- Multiple space-physics and scientific support experiments
- Soft landing on unknown surface terrains
- Manned capsules
- Minimum weight
- Launch time restraints (windows)

among many other scientific and engineering support functions.

Each of the above technical requirements plus customer schedules must be fully weighted in defining a total reliability program compatible with the system definition utilizing latest state of the art technologies.

Since total mission success of a system is desired, the basic concept developed was that any reliability effort should be systems oriented from start to finish. As a result, a total program consisting of four phases based on this concept was developed.

PHASE I

PROPOSAL AND PRELIMINARY RELIABILITY DESIGN

The first phase is initiated upon the receipt of the Request for Quotations (RFQ). This phase includes attendance at the bidders conference, technical reviews of referenced and required supporting specifications, etc., plus the formulation of various spacecraft systems, configurations and operational criteria to accomplish the mission. During this time the primary reliability tasks include:

- Developing initial system reliability block diagrams
- Determining initial reliability apportionment
- Formulating initial math model
- Establishing initial redundancy tradeoff studies
- Establishing level of materials and parts reliability and quality requirements
- Participating in vendor review teams and preparing subcontractor reliability requirements, specifications and tasks for statement of work inclusion - especially defined time required deliverables.
- Participating in system conceptual design reviews. Including component and materials consultants where new and different space environments may be suspected to have an effect upon existing systems.
- Based upon initial systems definition, parts count estimate, and preliminary reliability studies, formulating an initial A Priori Reliability prediction.

Up to this point, the primary inputs to Reliability are estimates based upon experience in other programs, standard known and published failure data, potential "state of the art" improvements as a result of research done by the company, the industry, and governmental agencies. This first approximation of system capability to meet the reliability objectives will also provide a guide as to where the quality assurance program's major effort should be directed to protect inherent reliability. Considerable flexibility is required during this phase to utilize and optimize all data inputs, which are normally quite meager and lacking necessary controls for suitable reliability consideration. As the information starts to increase, measurement of reliability growth becomes possible and the second phase of programming and definition is reached.

PHASE II

DETAILED RELIABILITY DESIGN

Phase II of this type of program effort overlaps Phase I and may parallel many of the tasks for a period of time. It is during this time span, however, when first circuit and subsystem designs are developed, reviewed and the physical detailing of the reliability model is accomplished.

This includes:

- Developing a detailed reliability program plan for customer approval
- Refining the reliability model based upon design reviews and early engineering information
- Developing the subsystem and unit math models and reliability apportionments
- Defining reliability apportionments in formal released reliability system specifications
- Establishing the necessary requirements and constraints for space usage qualification of hi-reliability components, materials and processes
- Implementing verification and traceability of parts
- Qualification testing
- In-process sampling
- Failure review activation and formal failure reporting; initiating analysis and corrective action
- Design review initiation by means of failure mode analysis and participation as a formal member of all design review committees
- Initiating subcontractor reliability surveillance
    - Subcontractor reliability requirements are detailed in the statement of work and subcontractor reliability specifications
    - Surveys in depth are made to acquaint the subcontractor with the requirements of spacecraft such as:
        - Reliability program plan
        - Reliability prediction
        - Growth curves

  - Failure reporting
  - Failure analysis
  - Failure mode audit
  - Periodic status reporting
  - Coordination meetings
- Motivation training

This program is initiated at subcontractors during this phase, and continues commensurate to degree of response. The training program should stress a basic understanding of the particular spacecraft requirements, via visual aids such as slides, charts, films, etc., related to the subcontractor products.

Phase II is a critical time sequence ensuring from a reliability management aspect that essential tasks and standards are being integrated into all of the required program elements. It is at this time that a critical analysis of all reliability operations in being are scheduled, to ensure that each is clearly defined to avoid ambiguities and assume desired results. It is also in this phase that initial system prediction commence, and are continually upgraded via test data and failure analysis as available from engineering models, prototypes and subcontractors reports. Due to the nature of such data on early developmental hardware, system design, etc., large dependence must be placed upon failure mode analysis, A Priori estimates and historical information to supplement the above test data. It is in Phase III where the transformation to hard engineering data and tests results on delivered flight hardware and systems test results begin to provide reliability growth information and allow confidence limits to be placed on predictions supplied to management and the customer.

PHASE III

SPACECRAFT FABRICATION, TEST AND OPERATIONS

In this phase of the program, engineering data begins to become available from breadboard, prototype and type approval tests being evaluated under simulated launch and space environments. Type approval testing of the initial subsystem, system, prototype spacecraft, as well as various related system models - structural, thermal control and propulsion - will be providing time, cycle, performance and failure data. The subcontractors' qualification programs are in process and reliability growth information as a result of test conditions becomes available for management information.. Here, therefore, the program includes:

- Based upon test results, refined failure mode analysis, and failure reporting, reliability predictions and math model are refined.
- Failure review board follows closely failure reports, diagnosis and originates corrective action requests.
- Failure mode audits are updated and completed.
- Close management surveillance to subcontractors is provided, and ensures that each is producing contractual outputs.
- Reliability assurance test programs on flight assured hardware are developed. These programs are based upon an "Equivalent Mission Cycle" (EMC) concept and require rigorous data review of all program testing as well as a commitment of basic flight hardware for unit and subsystem mission simulation evaluation. Senior reliability personnel are designated as "Test Directors" and initiate planning, scheduling and reliability test operations.
- Subcontractor reliability program milestones are reviewed. Submittals such as failure mode audits, monthly predictions, growth curves and data submittals are analyzed and on-site surveillance trips are increased.
- Reliability training and motivation programs are prepared and presented to prime and subcontractor personnel at key hardware delivery milestones, to maintain personnel awareness of the role each plays in their product reliability.
- As spacecraft systems become available for testing, system test teams are activated and reliability test team representatives are assigned from initial test to field launch site to ensure that all test data include timing and cycle information, that adequate information is supplied on failure reports, and that corrective actions on system problems, test equipment and procedures are initiated and followed-up. Adequacy of subsystem and unit data packages through a reliability unit history log is also a prime responsibility.

- The effect of all unit and system specification and engineering changes are reviewed and approved by Reliability as are all specifications and procedures initially released.

- Reliability launch critiques are prepared during this phase for day by day updating by the test team reliability member. This includes status of all units with respect to end of life, changes in characteristics or performance, on-off cycle wear and system tradeoff where system redundancies are involved. Of particular significance would be parameter drift indications during final "Operational Readiness" testing.

- Finally, as each spacecraft system is assembled, tested, shipped and mated to its launch vehicle, a continuously updated reliability prediction for spacecraft performance to its design specification is maintained and provided to management for launch visibility.

Phase III is the hardware and launch oriented portion of these programs and extends from source surveillance to launch operations. It uses a priori estimates, available test data, failure mode analysis, failure reporting, systems test data feedback, special reliability test programming, machine techniques and analysis to achieve continuously updated reliability data and prelaunch predictions. This will continue to overlap Phase IV until last program launch when only postlaunch analysis and final reporting will be major milestones.

PHASE IV

POSTLAUNCH ANALYSIS

Reliability effort provides one of the strongest inputs to postlaunch analysis. The responsibility for failure review, system data package, system test failure reporting and launch operations data action accountability lie within the normal scope of work. Working in consonance with the Mission Analysis team, the reliability organization has a responsibility to provide a number of detailed and specific inputs to the program and to launch and program reporting.

These include:

- Support of the postlaunch data analysis team is provided via review and analysis of prelaunch, launch and operations data and problem summaries.
- As a result of analysis, corrective action for next launch is initiated to update requirements and spacecraft hardware or procedures.
- Management is provided with postflight reliability summaries and recommendations for next flight action.
- Based upon previous postlaunch analysis and corrective actions, the next launch reliability critique and prediction is refined, communicated to management, and incorporated in the launch director's reliability critique manual.
- Upon program launch completion, a summary reliability report is prepared briefly describing program achievements and providing a full accountability of system, unit and component performance and potential solutions to next generation problems.

PROGRAM ORGANIZATION

Each spacecraft reliability office is uniquely associated line-management wise with its Project Management Office. Reliability is considered a system function associated via System Engineering to Project Management and hence Division Management. In the same way, therefore, that System Engineering defines and controls system performance, Reliability defines and controls system reliability. Following are some of the reliability oriented controls that are specifically for space applications:

- Space qualified materials
- Space qualified preferred parts
- Space qualified processes
- Cradle to grave parts accountability
- Variation and substitution authorization cognizance
- Approval of all specifications and procedures
- Approval of all specification and procedure changes
- Subcontractor reliability program control
- Chair failure review boards
- Failure reporting and failed parts collection
- Equipment reliability history log
- Reliability representatives on all system test teams
- Reliability assurance test program direction
- Postlaunch analysis team representatives

In each of these facets, the reliability team supports Project Management in making appropriate decisions concerning spacecraft COST, SCHEDULES, PERFORMANCE and RELIABILITY. By including Reliability in fourth place, due consideration is being given to the risk decision that Project Management must make concerning the four major elements. Where Reliability may be only a minor factor, certain decisions may be made with respect to a launch of an unmanned spacecraft that would have a different connotation for a manned launch, i.e., one of the number of scientific instruments inoperative might not hold up an unmanned launch, but any system primary or secondary being inoperative would hold a manned launch.

RECOMMENDATIONS

The following progression of reliability milestones are recommendations to achieve program objectives which will readily integrate with most management structures, operating policies and directives:

- Establish an effective Systems Reliability Capability at program inception. Major program commitments occur during this phase.
- Insure that all potential subcontractor surveys include adequate Reliability and Quality Assurance team members and that their survey weighting is significant.
- Develop the system reliability specification and release with program go-ahead to ensure design cognizance of hardware objectives.
- Establish immediately the necessary parameters for parts, materials and processes qualification and control. Implement the quality program and establish space qualified parts and materials handbooks for the program in sufficient time to influence the design.
- Establish the requirement for "Reliability Growth Curves" for first breadboard performance through delivered flight hardware to assess early potential problem areas. This is particularly true with subcontractors.
- Establish as an effective program requirement a Reliability Assurance Test Program, suitably designed to provide confidence in the hardware-mission sequence capability. This entails assurance that all test data is available, failure reporting is firmly established and flight assured hardware is committed to a special test program to supplement all other program data - normally insufficient to project desired confidence in design objectives.
- Provide an early and continuous company and subcontractor reliability training and motivation program to ensure that all program associated personnel understand the importance of their individual efforts and potential damage they can cause through not doing their best.
- Maintain tight surveillance over all subcontractors' efforts, insuring that milestones are met, failure analyses are timely and corrective actions initiated, that program objectives are thoroughly understood and competently implemented.
- Maintain a close liaison with the customer's reliability interface to insure that there is awareness of problems - "NO SURPRISES" - and that he, the customer, in turn, can easily and for the most part be informally advised of related areas that might influence the subject program or be a potential problem. Such a relationship is extremely important to achieve, and yet may be one of the most difficult to cultivate and maintain. Maximum effort here to obtain a working interface will certainly pay off in program achievement and customer understanding and concurrence.

CONCLUSION

The discussed four-phased program provides the necessary ingredients to satisfy most program requirements and, while accepting customer requirements, will retain the individual characteristics of any space oriented organization. Each contracting organization will have its own specifications and programs which must be adhered to, but each contractor must still retain the individualism of organizational integrity and management structure that allows for economical and timely execution of contractual commitments. Such phasing provides clear identifiable program benchmarks and allows project management visibility into problems and progress.

MAN-RATING THE GEMINI-TITAN II AIR FORCE LAUNCH VEHICLE

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Summary

The paper examines the disciplines and new approaches being utilized which together comprise the "Man-Rated" Gemini-Titan II Air Force Launch Vehicle Program. It discusses the critical component program; the technical and specialized skill training and certification program; the Quality Systems Engineering approach; the advanced corrective action and failed parts analysis program; the classification of characteristics program established for fabrication and test; the workmanship discipline approach; individual performance measurement; the Zero Defects program; personnel motivation; Air Force Space Systems Division disciplines imposed by the "Pilot Safety Program"; the limited life component program; the procurement controls imposed; and finally, an evaluation of the effectiveness of this "Man-Rated" approach.

Man-Rating in Design

Man-rating is defined as the marshalling of the disciplines necessary to achieve a satisfactory probability of crew (astronaut) survival. Technically, this probability may be expressed in terms of reliability of the launch vehicle and reliability of the Malfunction Detection and Escape System. Mathematically, these terms are linked in the equation -

$$P_{CS} = R_{LV} + (1-R_{LV})\, R_{MD \text{ and } E}$$

where:

P_{CS} = Probability of crew survival

R_{LV} = Reliability of launch vehicle

$R_{MD \text{ and } E}$ = Reliability of the Malfunction Detection and Escape System.

It can be seen that to maximize probability of crew survival, careful attention must be given to such launch vehicle items as:

(1) Use of redundancy to improve launch vehicle reliability.
(2) Design of a reliable Malfunction Detection System based on analysis of launch vehicle failure modes.
(3) Utilization of the crew as part of the Malfunction Detection System.
(4) Designing a vehicle checkout program which minimizes the probability of launching a defective vehicle.
(5) Utilization of a disciplined Quality & Reliability Assurance Program to prevent degradation of inherent design reliability.

Malfunction Detection System

The MDS monitors launch vehicle parameters that affect crew safety. It actuates lights and meters in the spacecraft which, in display form, warn the astronauts of potential dangers or malfunctions. The Gemini MDS was designed to make maximum use of the astronaut's decision-making capabilities which resulted in a system of increased reliability and simplicity. It is a manual abort system whereby the astronauts, from the displayed performance parameters, determine if abort action is required. The choice of performance parameters was based on careful analysis of modes of failure, probability of these failures to occur, times to catastrophe, and crew reaction time at various phases of the boost flight. The parameters selected for display are:

Engine thrust chamber pressures for both vehicle stages,
Tank pressures on all tanks in vehicle,
Overrates in pitch, yaw and roll,
Staging sequence,
Switchover from primary to secondary flight control system.

The MDS was designed with full sensor redundancy, redundant power supply circuits, separated installations of primary and secondary system elements, and fail safe design and hook-up to minimize possibility of signalling a false malfunction.

Flight Control System

During the course of conducting the failure mode analyses, one particularly critical type of failure evolved - that of an engine "hard-over" during Stage I flight approaching "Max Q". At this point, structural break-up would occur in approximately one second. This is obviously too rapid for the manual abort philosophy which requires time for detection, reaction, escape initiation and actual escape. It was determined that a way out of the dilemma was to radically reduce the likelihood of a "hard-over" engine condition by adding a redundant flight control system to the launch vehicle.

The redundant system utilizes a primary string of a three-axis reference system (with integral programmer), a radio guidance system, an autopilot, rate gyros, and a hydraulic system feeding a newly designed tandem or dual actuator. The secondary system utilizes the inertial guidance system of the spacecraft, a secondary autopilot, secondary rate gyros and an independent secondary hydraulic system feeding the alternate side of the tandem actuator.

Switchover is automatically accomplished if any of the following malfunctions occur during Stage I flight.

(1) Loss of hydraulic pressure in the primary hydraulic system.
(2) Engine thrust chamber movement to "hard-over" position.
(3) Excessive turning rates in any of the three (3) axes.
(4) Manual command of crew.

Since dynamic pressures are much reduced during Stage II flight, the "hard-over" condition is no longer critical then. As a result, no redundant hydraulic or tandem actuators are used in Stage II. However, switchover of the remainder of the system is automatically accomplished if turning rates are excessive in pitch, yaw or roll or by manual selection of the crew.

Electrical System

The electrical system was designed with redundancies in all major circuits. There are two separate power buses fed by individual batteries. These alternate supplies are fed to such components as:

(1) Program initiate relays
(2) 40 second time delay relays (lockout spacecraft shutdown)
(3) Staging control relays
(4) Staging switches
(5) Explosive separation nuts
(6) Stage I & II engine shutdown circuits
(7) Guidance shutdown relay for Stage II
(8) Manual booster shutdown switches
(9) Range safety equipment

Critical Component Program

An important element in the Gemini Quality program was the establishment of a critical component program to prevent degradation of inherent reliability by means of "tender loving care." A review of the airborne components was made, and utilizing the selection criteria shown below, 54 items were designated as critical.

(1) failure modes associated with the component design and application will result in vehicle mission failure and crew abort.

(2) failure modes associated with the component design and application will significantly decrease the probability of mission success.

(3) failure of the component during countdown will result in a hold or kill and prevent meeting launch window required for rendezvous mission.

(4) The component is a design unique to Gemini and has no previous manned vehicle application.

The selection criteria were also supplied to the manufacturers of the 54 critical components with the requirement to review and select the piece parts of the component which should be classified as critical. These piece parts lists were then reviewed, negotiated and finally approved by the Martin Reliability Group and the Air Force. The following controls were then instituted on the resulting hundreds of piece parts.

(1) Identify critical parts with tags, decals, etc.
(2) Segregate critical parts so that they can be more carefully handled.
(3) Institute special MRB (Material Review Board) procedures to carefully control repair or rework of critical parts.
(4) Prohibit acceptance of parts which have exhibited transient malfunctions unless a conclusive reason and conclusive repair can be found.
(5) Control changes to design and manufacturing processes to maintain qualification configuration.
(6) Provide planned storage and handling to minimize damage.
(7) Provide prompt, detailed failure analysis and corrective action on failed critical parts, but obtain Martin permission for corrective action affecting design or processes since this constitutes a configuration change.
(8) Indoctrinate personnel handling critical parts in the above requirements.

(9) Source inspect critical components, utilizing resident Martin Quality personnel to audit for adherence to critical part requirements.
(10) Provide certificate of compliance with above requirements with each shipment of critical components.

The above controls have also been instituted on components fabricated within Martin and the entire program has resulted in an awareness of the special requirements of the Gemini Program. The number of trouble reports written on critical components at the launch site, for example, reflects very little difficulty with critical parts.

Training & Certification

Selection of Personnel

A prime objective in the selection of Gemini personnel was utilization of Titan experience. This was especially true in design, vehicle testing, fabrication and launch crew operations. The initial selection, therefore, factored in a group of highly trained and experienced personnel. This group was supplemented by additional qualified people and then all the personnel were processed through a carefully planned series of orientation, safety and technical training courses suited to the particular individual and his planned assignment. The courses culminated in an evaluation of the absorption of training by means of test scores and of performance on-the-job. The end result is to have every Gemini team member technically competent to perform his job, have him understand the importance of his contribution to the Gemini program, and the importance of the objectives of the Program to the country.

Certification cards are awarded to each trained man on the Gemini team. Quality personnel make certain that only "card carrying members" are performing Gemini tasks.

A special on-the-job training program exists at the launch site because of the requirements to work together as a launch team. Special training exercises have been run in loading propellants, simulating hazardous operations or safety problems. Crew certification has resulted, in addition to individual certification, and this program has paid handsome dividends in the form of smooth, disciplined and coordinated countdown exercises.

Certification boards, consisting of managers of operating divisions, administer the certification program and maintain the high standards demanded by the program.

Motivation

Much emphasis has been placed towards developing the attitude of each member of the Gemini team towards his job. The requirements of the highest standards of workmanship, the need for striving towards perfection, the development of pride in workmanship have been stressed utilizing astronaut visits; special Gemini stickers on equipments, decals for automobiles of team members, special Gemini pins; imaginative posters, coveralls or shop coats with Gemini insignia, and a constant program of articles using the Martin newspaper are a few of the motivation techniques being utilized.

An additional, intensive motivational campaign has been instituted in the form of "Zero Defects". This program instituted a series of defect goals, which become more stringent with time, and against which measured defects are plotted. The program embraces all phases of activity including non-hardware support areas. Award programs have begun, both on an individual and departmental basis. The main point of the entire campaign is the constant reminder that there are no "insignificant mistakes" on the Gemini Man-Rated Program.

The trouble reporting system has been modified to print out on an automated tab run the badge numbers of individuals contributing to operator errors resulting in hardware discrepancies. This tab run aids in determining additional training requirements or, in extreme cases, de-certification of Gemini team members.

Limited Life Program

An operating life test program begun on the Titan program was continued on the Gemini program. In general, life test data was used in conjunction with qualification test data, together with mode of failure analysis and failure histories to arrive at a maximum operating life limit for selected time sensitive components. These limits are then used as a basis for rejection if actual operating hours exceed the limit established. An automated equipment time recording system is in operation which starts to operate during acceptance testing at the supplier's facility, continues through receiving inspection and bench testing at Martin, through sub-system, system and combined systems testing in the factory and at the launch site. System operating times are obtained from specially designed multi-channel recorders. The tab run prints out by part number, and by serial number, the actual and accumulated operating hours with identification to various test phases in which the time was accumulated.

A refurbishment plan is under negotiation which allows for increases in operating life based upon careful replacement of selected time or wear sensitive piece parts, followed by a retesting program.

The limits that were selected are conservative and in consonance with the Gemini reliability goals.

Quality Systems Engineering

A Quality Systems Engineering Group was established on Gemini by assigning competent professional engineers to the various sub-systems of the vehicle at the beginning of the design effort. These engineers participated in drawing reviews, specification preparations, vendor selections, qualification and acceptance test procedure reviews, design reviews, etc. In particular, they were assigned the following major tasks:

(1) Maintain a continuous study of assigned system to permit technically competent decisions in following tasks.
(2) Review specifications (contract, test, procurement) for addition of Quality Assurance provisions.
(3) Review test, maintenance and operating procedures for adherence to specifications for adequacy and for selection of key check points.
(4) Review and approve Qualification, Acceptance and Production Monitoring test procedures and reports from suppliers and for in-house testing.
(5) Examine and approve specification deviations and waivers, Material Review Board reports and minor variation review reports. This was done to increase the technical level in decision-making associated

with non-conformances.

(6) Review engineering changes with an emphasis on establishing retrofit inspection and test plans to restore hardware to its pre-change integrity.

(7) Participate in test team operation to review adequacy of selection of key check points (item 3 above) and to perform on-the-spot failure analysis on test anomalies.

(8) Analyze and accept for Quality, test results, telemetry data, land line recordings, etc.

This group is currently performing the bulk of the Corrective Action problem investigations and supplying technical liaison with the team of resident field inspectors.

Quality Classification of Characteristics Program

Because of the utilization of existing Titan engineering as a base for the Gemini design and since Titan did not have a classification of characteristics program, the Quality Engineering Group on Gemini carried out an informal classification on fabrication process plans and test procedures. Utilizing criteria established for this specific purpose, critical characteristics were highlighted for special attention and data documentation by line inspection personnel. These special requirements were integrated into the manufacturing planning so that they were highlighted also for the fabricators and testers as well.

Procurement Controls

In the early stages of the design effort, a special set of procurement specifications were prepared which covered reliability requirements such as trouble reporting, equipment operating time recording, failure analyses, failed parts analyses, design reviews; critical component requirements; data and documentation requirements such as format and content needs of qualification and acceptance test procedures and reports. These specifications were referenced as applicable in the individual procurement drawings and the extent of these references reviewed by Reliability and Quality Engineering.

In placement of business with new suppliers, bid evaluation was performed by Design Engineering, Reliability, Quality Engineering and Procurement.

Acceptance and Qualification procedures prepared by the suppliers were reviewed and approved by Martin before testing could proceed. Test Reports were then reviewed and accepted similarly.

Detailed contractor acceptance plans were prepared by Quality Engineering for use by resident Quality personnel. These are inspection instructions embracing process reviews, quality system audits, hardware audits, specification compliance requirements and test procedure data.

Vendor workshops were planned and carried out during which the special requirements of the Gemini Program were explained and discussed. Interface problems were aired and corrective action instituted as required. These sessions served greatly in bringing these suppliers into the Gemini team with increased enthusiasm, cooperation and understanding.

A continuous system of audits in suppliers houses is serving as an impetus to keep the rigid controls in active force.

Vehicle Chaperones

To assure continuous compliance with contract, specifications and drawings, a Quality Assurance expert is assigned to each launch vehicle as chaperone. He begins with the tank fabrication at Martin-Denver, conducts the tank roll-out inspection there, travels with the tanks to Martin-Baltimore, remains with the launch vehicle through final assembly and vertical tests, is responsible for acceptance of vehicle documentation during the Acceptance Team exercise at completion of vertical tests, accompanies the vehicle to Cape Kennedy, observes stowage of the vehicle during flight, verifies loading and unloading operations, participates in the receiving inspection at the Cape and observes the flight test preparations. He maintains a historical record of significant problems, their applicability to his vehicle and their resolution. Since he is the one individual who devotes full time to a particular vehicle, he is best equipped to have readily available the major problems that have occurred during fabrication, inspection, shipment and testing.

Pilot Safety Program

The major function of the Air Force Space Systems Division/Aerospace Pilot Safety Program is to ensure that the concern for crew safety is contained in launch vehicle planning, is reflected in appropriate activities, and that the vehicle history is well documented and thoroughly assessed prior to launch. It is implemented by a continuous technical monitoring of all phases of the program from design through launch. It is also implemented by concentrated reviews at key focal points in the build-test-accept sequence. The areas that are rigorously reviewed are:

(1) Engineering design
(2) Development testing
(3) Engineering changes
(4) Production plans and processes
(5) Inspection
(6) Testing
(7) Handling
(8) Acceptance activities
(9) Launch operations
(10) Configuration documentation
(11) Test data and procedural reviews
(12) Corrective action status and problem resolutions.

Stringent requirements are imposed during the acceptance phase and hardware is not accepted until Air Force Space Systems Division and Aerospace are convinced that the hardware and documentation comply with specifications and contractual requirements. This acceptance is characterized by a methodical, uncompromising attitude.

Roll-Out Inspection

This is the first major inspection conducted upon completion of tank fabrication at Martin-Denver. The Air Force review covers tank weld information, significant problem history, laboratory tests on materials, stress analyses on weld repairs or discrepancies, and a "walk-around" inspection to make certain there are no doubts as to the man-rated quality of the vehicle tanks.

Vehicle Acceptance

Upon completion of all sub-system and system testing in Martin-Baltimore, a Vertical Acceptance Team conducts a comprehensive review of hardware logs, test results, observes combined systems test and audits the assembled hardware. The team assures that the following criteria has been met:

(1) Completion of satisfactory component, sub-system, system and all systems acceptance test.
(2) Completion of documentation of hardware status including history and identification of components by serial number, test data, configuration status and problem areas encountered, together with their resolution.
(3) Special detailed report containing critical component history.
(4) Completion review (no vehicle or spares shortages).

Flight Safety Review

A similar review is conducted by the Flight Safety Review Board just prior to flight and constitutes the final activity of the Pilot Safety Program by making the final decision of committing the booster to launch.

This review covers the following areas:

(1) Significant details of previous Gemini and Titan launches.
(2) Modifications to hardware, mission and procedures since previous review.
(3) History of launch vehicle through production, acceptance and AMR activities.
(4) Report and recommendations of the Launch Vehicle Status Review Board.
(5) Complete technical review of the significant problem areas.
(6) Report on personnel changes of the launch crew and status of flight readiness teams.

Following the above review, the FSRB decides whether or not to commit the vehicle to launch and presents this decision to the senior NASA representative for his concurrence or rejection.

Corrective Action and Failure Analysis Program

In adding man-rating to the trouble reporting - corrective action loop, the most important changes are:

(1) Generation and utilization of trouble report data from all possible applicable sources.
(2) A problem investigation of all actual and potential problems, with emphasis on striking the words "random failure" from the vocabulary.
(3) A planned corrective action loop embracing Martin-Denver, AMR and all Gemini suppliers which directs necessary corrective action in careful detail with authority to carry it through.
(4) Detailed problem documentation and disposition decisions for each trouble report. Note that each report and problem is reviewed and the disposition concurred with by SSD/Aerospace.

(5) Failed parts analyses carried to sufficient depth by experts in this technique, utilizing new non-destructive testing techniques to firmly establish mechanism and failure cause. This includes dissection of piece parts such as diodes, capacitors, potted modules, etc.

Because of similarity of design, Titan sources of data are utilized to a great degree.

Open problems are added as constraints to significant PERT events and hence get top management attention and pressure for resolution.

Speed of response of suppliers in carrying out failed parts analyses is a major problem in maintaining rapidity of response to open problems. It should be noted that all open problems must be reviewed and a determination made whether or not to proceed with the next key PERT event. It is especially important that the number of open problems be small prior to a Flight Safety Review Board or a Vehicle Acceptance Team Meeting.

Evaluation of Benefits Gained from Man-Rated Program

Many benefits of the above program are starting to appear quite clearly. Some of these are:

(1) The Gemini-Titan II Launch Vehicle will have all components fully qualified prior to flight.
(2) The number of hardware failures on airborne components at AMR is significantly low.
(3) The difficulties encountered in testing procedures, ground equipment interfaces, facility interfaces, were significantly low for the first launch vehicle.
(4) Test schedules were set and could be adhered to because of comparative freedom from major problems.
(5) EEI (Electrical/Electronic Interference) problems have been at a minimum.
(6) Air Force confidence has been high because of depth of response and scope of problem coverage.
(7) Marked inroads in area of vendor control have been pioneered on this program with resulting higher reliability hardware.
(8) An unprecedented position of value and contribution to success of the program has been achieved by the Quality and Reliability Groups.
(9) Full spares support of qualified hardware was available at AMR to support the first launch vehicle on schedule.
(10) A complement of highly trained and motivated personnel are available to continue the efforts accomplished to date.
(11) Interfaces with associate contractors on the Program have been firmly established and cooperation is excellent, particularly in the technical and trouble reporting - corrective action areas.

In conclusion, although at the time of this writing the proof of performance is yet to come, it is felt that the enormous increase of attention to details, the "tender loving care", the many disciplines added, the competence of personnel, etc., will all contribute to a program which may well serve as a model for future manned space efforts.

References:

ER 12521-15 - "Man-Rating the Gemini Launch Vehicle".

ER 12254C - "Gemini Quality Assurance Plan".

SYSTEM RELIABILITY TESTING IN SPACE SIMULATORS

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ABSTRACT

All too often the cost in time and money seriously limits the required testing programs. Scientific personnel accept the challenge of space craft design but are rarely stimulated to design and operate the required testing facilities. The space craft test engineer of today must, of necessity, be an experienced expert in all phases of space technology power-plants, solar arrays, hydraulics, pneumatics, structures, dynamics, ultra-high vacuum, solar effects, etc.

Today's space environmental testing procedures and techniques have developed a number of varied and specialized test facilities to insure system reliability of space craft. Of these many types, the most important and essential are the Orbital Simulators. The urgency of our current space efforts has been responsible for great strides in high vacuum equipment techniques and systems used for space simulation.

To review briefly the state-of-the-art and to provide general familiarization with the reliability testing theory, methods and application of the Orbital Simulator for space flights will be presented.

To make an over-all system reliability test, just before the launch--i.e., to find out exactly how the system will behave in orbit--space simulators are used.

INTRODUCTION

Man, on the threshold of free space flight travel, is realizing today a dream and a challenge that have been present from the earliest recorded histories of time. The question of what lies out in the "heavens" will soon be answered by highly trained astronauts in advanced design space vehicles. However, before man can safely undertake such an adventure, numerous unmanned exploratory flights are required. These flights are expensive and failures cannot be tolerated. It has been well said, "We should not build junk yards in space." It is highly desirable to be sure in advance that the probability of a successful flight is very high. To achieve this objective requires mastery of almost endless problems and apparently insurmountable obstacles.

Today's space craft consists of a highly complex integration of intricate subsystems: guidance, propulsion, communication, power, etc. Not only must each subsystem be tested to insure proper design, but the complete system must be tested when subjected to the expected combined environments. The old philosophy that a chain is no stronger than its weakest link is a principal axiom of today's reliability testing engineer. The space simulator is a principal tool of the reliability testing engineer. The simulator must be equipped with checkout equipment which in general is more advanced and of a higher operational reliability than the space vehicle to be tested.

SPACE SIMULATORS - THE OBJECTIVES

There are numerous environments in true space, such as ultra high vacuum, temperature, radiation, gravity (or lack of it), meteoric showers, magnetic fields, etc. It is generally accepted that for space vehicles greater than four feet in diameter reproduction of ultra high vacuum and temperature is the most feasible approach. Unfortunately, the elimination and control of the gravitational field is beyond the existing "state-of-the-art" of a space simulator. However, the surfaces of the test specimen can be subjected separately to nuclear radiation, meteoric dust, solar radiation, etc., to determine their effects.

The principal objectives of the space simulator chamber while subjecting the test specimen to the combined environments of high vacuum and heat flux are:

1. To obtain test data to assist in the determination of the Mean-Time-To-Failure and the principal failure modes of the space satellite.

2. To establish the degree of thermal compatibility of the test vehicle and its subsystems.

3. To determine the degradation of equipment when signals or voltages of the satellite have been deliberately changed.

4. To determine the degree and effects of outgassing and redeposition.

At the time of this writing, complete solar simulation is not feasible for a large test specimen for proper illumination, spectrum distribution, uniformity, and intensity have not been accomplished. The approach used at the Lockheed Missiles and Space Company at Sunnyvale, California, in the High Vacuum Orbital Simulator (HIVOS) is to provide a dependable, long-duration combined environment of high vacuum and a controllable heat flux. The heat flux is capable of providing both free space and earth orbital thermodynamic conditions.

SPACE SIMULATORS: DESIGN AND CAPABILITY

The HIVOS high-vacuum orbital simulator (1962) is Lockheed's second generation space chamber. The "forefather," named High Altitude Temperature Simulator (HATS), was built in 1958. This chamber pioneered simulator technology and introduced into industry the first orbital simulator. The then existing vacuum technology state-of-the-art limited the vacuum to approximate only a 100-mile altitude and simulated the temperatures with radiant heating and refrigerating systems.

The HATS chamber (Figure 1) is constructed of stainless steel, 8 feet in diameter and 9 feet high. The bottom of the chamber forms a door which is raised and lowered by a 10-½ inch diameter hydraulic lift capable of lifting 4000 lbs. The pumping system consists of three diffusion pumps, a booster diffusion pump, a mechanical roughing pump. Use of liquid nitrogen-filled cold traps reduce the possibility of oil vapor from the vacuum pumps backstreaming into the chamber. Temperature control during tests is obtained by using radiant and refrigeration systems within the chamber. There are 14 separate sections; each uses Freon 13 and Freon 22, individually controlled.

Utilizing the experience gained in the design and operation of the HATS facility, Lockheed Missile and Space Company specified the operating parameters for the HIVOS facility (Figures 2,3, and 4).

The HIVOS facility offers the following minimum capabilities:

1. High Vacuum (10^{-8} Torr range) continuously for periods up to one year

2. Heat flux programmable to simulate noon to twilight orbits and even has reserve capacity to exceed the solar radiation experienced at Venus

3. Cold-wall heat sink to near liquid nitrogen temperature (-293°F)

4. Automatic synchronous programming of test specimen operation

5. Random access programming of test specimen

6. Automatic data acquisition

7. Complete data processing facilities, including access to complex computers such as the IBM 7090 which are available in the Sunnyvale Complex.

The HIVOS chamber measures 18 feet in diameter and 20 feet in length; it is bottom loading with a liquid nitrogen filled cold-wall composed of interlaced extruded aluminum chevrons. The chevron surfaces facing the center of the chamber (Figure 5) have been sand blasted and anodized dark grey to improve their absorption of infrared radiation; the surfaces facing the inner walls of the chamber are polished to reflect radiated heat. Cold walls cover all the inside surfaces of the chamber (Figure 6). Liquid nitrogen at -320°F circulates through the hollow cold wall and removes all heat absorbed by the plates. The gaseous nitrogen which is boiled off by the absorbed heat is separated and vented to the atmosphere; the remaining liquid is returned to the storage tanks for reuse. Approximately 12,000 gallons are used each day when normal orbital simulation is programmed. There is a total of 2850 square feet of cold-wall surface area. The Heat Flux Simulator (Figures 7 and 8) consists of clear quartz lamps with tungsten filaments and polished parabolic liquid-cooled aluminum reflectors. The lamp reflector assembly is designed to produce an even flux pattern over the specimen. Twenty-six separately controllable zones are used. Special radiometers measure the radiant heat flux programmed to the test specimen. A digital programmer (Figure 9) is used to provide any heat flux which may be required under orbital conditions: i.e., polar, equatorial, noon and twilight. The maximum output of the heat flux simulator is 800 BTU/ft^2/hour. The minimum output is less than 5 BTU/ft^2/hour. An automatic control panel (Figure 10) controls the operation of the twenty-two 36-inch liquid nitrogen trapped diffusion pumps, the four diffusion ejector pumps, and the four mechanical roughing pumps.

The chamber also has the capability of self-cleaning by bakeout, under vacuum, at a temperature of 400°F. This is accomplished by circulating special fluid through a grid of over two miles of stainless steel tubing which is attached to the outside of the chamber (Figure 4). A special bake-out chamber closure with a network of heating tubes attached is used instead of the normal test closure. The building and pumping system were designed to permit the chamber length to be extended to 35 feet by removing the top of the chamber, installing an 18-foot diameter circular section 15 feet high, and rewelding the top section into place.

The existing pumping system is capable of producing a 10^{-6} Torr pressure in 90 minutes; a 10^{-8} Torr pressure in 8 hours. A pressure of 1×10^{-6} Torr can be maintained with a calibrated leak of 150,000 micro-liters per second.

SPACE SIMULATORS: THEORY AND SPACE ENVIRONMENT

Satellites and space craft travel through space vacuum, except for the short duration of the ascent and re-entry phases; thus the only significant external mode of heat transfer is radiation. Within the vehicles, however, heat transfer takes place by radiation, conduction and, in pressurized vehicles, by forced convection. For satellites, the external sources of energy are the sun (insolation) and the earth which serves both as an emitter (earth emission) and as a reflector (albedo) of solar energy. An external surface segment of an orbiting satellite is small enough to be considered isothermal.

The governing mathematical relationship, which is derived in the appendix, (Eq. 6) can be expressed as follows:

$$\sigma \epsilon T_W{}^4 = F_S S \alpha_S + F_R R \alpha_R + F_E E \alpha_E .$$

This mathematical relationship fully expresses the physical model involved in the simulation of orbital satellites in space flights. Equation 6 indicates that the temperature history of the outer wall depends only on the time history of the external radiation absorbed by the vehicle in space and its surface properties. It is evident, too, that it is not necessary to evaluate the complex internal heat transfer terms that appear in the appendix (Equations 1 and 2).

To provide necessary controls for the simulator individual segments, it is necessary to program variation T_w wall temperature at the rates of as indicated in Equation 6. In the HATS chamber this variation could be done by a series of cam curves to be used for control of orbital temperatures. Thus, cams could be generated with the following assumptions and considerations:

1. The orbital path is known

2. The orientation of the satellite is known

3. The satellite will receive radiation heat from two sources, i.e., earth and sun as presented above.

If the earth radiation and solar radiation are assumed constant, the total radiant energy upon any point of the satellite depends on two conditions: how far the satellite is from the source of radiation and how it is oriented to the space from the source. It is essential to make cams for the segment controls so that the segments are maintained as close as possible to the actual temperature profiles. It might be added that the cam shape is reflected by the temperature limitation and the temperature response of the segments. Where compromise is necessary, the cam shape can be modified by the total amount of the radiant energy actually emitted from the segments which must be the same as the theoretical values calculated for each orbit.

Space environment simulation for large specimens such as would require the services of the HIVOS chamber may be achieved at present by high vacuum and either of two types of radiation source. These are solar simulation and thermal flux simulation.

1. Solar Simulation: Complete solar simulation would include the entire electromagnetic radiation spectrum, but in current practice, consists only of the ultraviolet, visible, and infra-red wavelengths. This is roughly equivalent to wavelengths from 2000 Angstrom units to 100,000 Angstrom units, with the energy peak at 4600 Angstrom units (Figure 11, Spectrum Distribution). Approximately 8 per cent of the radiation is in the ultra-violet range.

2. HIVOS Thermal Flux Simulation: The HIVOS simulator uses type T3 clear quartz lamps with tungsten filaments as a radiation source. The resulting spectrum extends approximately from 3800 Angstrom units to 100,000 Angstrom units, with the energy peak at about 11,500 Angstrom units. There is no significant amount of radiation in the ultra-violet range; the major fraction is in the infra-red. By sequentially switching the lamps and operating a minimum number with increased power, the average wavelength can be reduced.

The simulated radiation affects the specimen in two ways:

1. Degradation of skin surface (primary characteristic of ultra violet): This is insignificant as far as penetration of a metal skin is concerned, but of great importance in regard to the skin coating.

2. Thermal effects (primary characteristic of infra-red): The thermal energy penetrates the skin or outer surface and is transmitted throughout the specimen.

Heat flux simulation in the HIVOS vacuum chamber is produced by using quartz lamps. These lamps have shown remarkable reliability (not one has failed in service during the previous 20-month operation). The initial cost as well as the operating and maintenance cost for a quartz lamp system is moderate compared to solar types. For these reasons, a large number of lamps can be installed within the chamber and controlled to provide a simulated orbit.

Solar simulation equipment is extremely difficult to operate within a vacuum chamber. In order to simulate orbital conditions in facilities employing the carbon arc or mercury-xenon equipment, rotation of the specimen is usually required, which can become a costly and complex operation, particularly if many "hard-line" electrical connections with the specimen are required. Efforts to program solar simulation energy about the specimen become even more involved, usually requiring a large number of chamber optical penetration ports, elaborate reflective devices, or both. Duplication of equipment is generally necessary for prolonged test operations because of the unreliable performance of carbon-arc and mercury-xenon lamps and the frequent servicing required. That these problems are considerable is evidenced by the fact that not one installation is known to be capable, in operation, of providing the continuous service required for long-term large-scale reliability testing programs.

Where solar radiation is required because thermal equivalents of the shorter wavelengths will not suffice, it is often practical to expose relatively small samples to carbon arc or mercury-xenon lamp radiation in a small vacuum chamber.

Since only surface effects are to be considered, the large and complex equipment used for thermal simulation is not required.

A common approach to the problem of space vehicle temperature control involves the use of thermal control surfaces, which may be used to perform the following functions:

1. To radiate heat generated internally in the vehicle.

2. To absorb solar radiation for the purpose of raising the internal temperature of the vehicle.

3. To perform both of the above functions simultaneously for different thermal requirements in several sections of the vehicle.

SPACE SIMULATOR: INSTRUMENTATION AND CONTROL SYSTEMS

The satellite vehicle being tested in HIVOS is under direct, precise control at all times during its simulated orbits.

The three automatic electronic systems test consoles located in the control room presently used for commanding the vehicle are:

1. Test Director's Console

2. Programming and Synchronizing Timer (PAST)

3. Automatic Programmer and Test Systems (APATS)

Test Director's Console

This is equipped to control and monitor vehicle electrical power including solar battery simulation. Its most important functions, however, are to monitor and, if necessary, override the vehicle's internal orbital programmer, and to send program commands to the vehicle through hardlines. Although the console was built by HIVOS personnel for a specific test vehicle, it was designed to be easily adapted to any program configuration and to provide any desired degree of monitoring and control of the test specimen.

Programming and Synchronizing Timer (PAST)

The PAST has three functions:

1. Time base generation

2. Orbit number and orbit time display

3. Vehicle program commands--storage and transmission.

The time base is common to all vehicle control and monitoring operations in HIVOS and ties together the several data collection systems. Based on a 16 KC clock, it is accurate to 0.001% per day.

The clock drives an orbital counter and timer which can be set to count simulated orbits with periods varying from 60 seconds to 999 minutes and 59 seconds. It can count up to 9999 orbits. The information is displayed for direct reading on two counters in the HIVOS control room and recorded in binary coded decimal form on oscillograph records.

The Programming and Synchronizing Timer can initiate predetermined time pulses which can trigger real time commands and stored program commands which are transmitted to the vehicle at the proper time during the simulated orbit. The tape reader for this function is located in the Test Director's Console, where it is easily accessible during orbital operations.

Automatic Programmer and Test System (APATS)

The APATS was designed and built for HIVOS to program, control, monitor, and record test vehicle and payload functions and performance during simulated orbital flight (Figure 12). It consists of fifteen racks of power, telemetry, programming, signal distribution, and recording equipment.

Programming. A punched tape reader provides the selection and timing of all comparison and evaluation measurements of the test specimen operation. It is capable of programming the selection of stimuli, selecting test points to be monitored, and determining program continuation in accordance with the measurements and evaluation results.

Control. Control of the test vehicle is through the UHF command link coaxial cable. Preprogrammed coded instructions are transmitted to the vehicle by the APATS telemetry console whenever it is commanded by the Program and Synchronizing Timer to initiate various orbital events.

Monitoring and Recording. The APATS contains recorders for the optimum handling of the several types of data obtained from the test vehicle in the HIVOS chamber, as follows:

1. A magnetic tape recorder

2. A high speed (20 points per second) paper tape printer for hardline vehicle data, such as temperatures, voltage, etc.

3. A low speed (5 points per second) paper tape printer. Any data channels can be selected and sampled, as described.

4. An oscilloscope and camera are available for photographically recording transient data which is of too short duration for a printer or oscillograph.

5. Important TLM calibration signals can be displayed on a digital voltmeter if required.

6. Four 20-channels event recorders are used for recording coded commands transmitted to the vehicle, both Real Time Commands (RTC's) and Stored Program Commands (SPC's).

7. Oscillographs are used for recording general vehicle performance parameters, either continuously for backup diagnostic purposes or periodically during selected simulated orbits for status monitoring. Four recorders provide 144 channels.

In addition to the several data recording devices incorporated in the APATS equipment, HIVOS capability includes a Low Speed Data System (LOSDAS). This equipment is basically a specimen temperature recording system which can sample thermocouples at the rate of two per second. The readings can be printed on either printed paper tape or punched cards or displayed for visual inspection. In addition, continuous analog data can be recorded on oscillographs. The equipment consists of two parallel 250 channel systems, one of which incorporates thermocouple reference junctions. The capacity of the LOSDAS is therefore 500 data channels, including 250 temperature measurements. A patch panel is used to select from 606 possible inputs from the test chamber.

THE ORBITAL SIMULATOR: TEST PROGRAMMING AND TEST CASE STUDIES

From experience, a practical, economical vehicle checkout and environmental test procedure has evolved. The procedure is divided into two parts, as follows:

Preparation

1. Test specimen assembled and subsystems checked

2. Instrumentation installed

3. Integrated systems test

4. Where possible, components are disassembled, cleaned, reassembled and installed on chamber door.

5. Chamber bake-out, if required, with blank chamber door

6. Final integrated systems test

7. Simulated orbital program during which all major events are programmed. Used as a final check for control data.

8. Test specimen installed in chamber and vacuum and data systems checked.

Operation

1. Simulate orbital program during which all major events are programmed. Used as a final check and control.

2. Pump down to minimum pressure (value depending upon the out-gassing rate of test vehicle).

3. Repeat simulated orbital program during which all major events are programmed.

4. Check thermal simulation system.

5. Conduct simulated orbit test.

6. Remove test specimen from chamber and repeat simulated orbital program for control data and degradation information.

7. Check subsystems as required.

8. Analyze data and prepare reports.

Agena Life Test--Case Study I

Recently a nine-month reliability testing program using the High Vacuum Orbital Simulator (HIVOS) was completed (Figure 13). This program included three 30-day tests and one 45-day test, subjecting the Agena flight vehicle to the combined environments of high vacuum and orbital temperatures. The vehicle was programmed as if in orbit. All vehicle subsystems, guidance, communication, payload, power supply, etc., were operated normally. Vehicle motion was simulated by the injection of error signals into the guidance loop. The response was carefully observed. Over 200 thermocouples were used to monitor and record the temperatures. The operation of the subsystem was monitored by strip chart recorders and automatic testing equipment which are part of the HIVOS Automatic Data Acquisition Equipment.

Degradation of equipment operation was determined. Catastrophic failures were also discovered. Necessary steps were taken to correct the weaknesses.

Fuel Dump--Case Study II

Fuel and other liquids must frequently be discharged from the vehicle into space. It is desirable that this discharge take place without imposing any unwanted torques or thrust to the vehicle. Also, the surface of the vehicle may be coated with solar cells or special thermodynamic finishes which are subject to serious degradation if exterior contamination occurs, as would occur, if these liquids contacted the surface. A test program was conducted in the HIVOS facility to determine if liquids similar to the Agena propellants (normally dumped upon completion of all ignition) would form jets with semi-vortex angles greater than 50°. If jets greater than 57° occur, then impingement on adjacent hardware transpires. Four different liquids were used, Methyl alcohol, water, Methylene chloride and Freon -300.

These liquids were selected because they would not contaminate the chamber and they have different freezing points and molecular weights. Complete photographic coverage was provided. The motion pictures definitely showed that for water the liquid jets had half-cone angles in excess of 57°.

During some of the tests a heated flat plate, simulating an adjacent main nozzle extension, had pressure transducers mounted. The pressures indicated by these transducers varied with changes of the plate surface temperature. This confirms the possibility that liquid droplets do impinge on the vehicle and the subsequent evaporation can produce vehicle perturbations.

During these tests liquid at the rate of one pound per second and a duration of 50 seconds was emitted into the chamber.

The actual pressure during the dumping of the fluid could not be measured because the vapor created erroneous readings of the ion gauges. Prior to the injection the pressure in the chamber was in the 10^{-6} Torr range and recovery occurred within minutes.

Thermal Stress of Agena Booster Adapter--Case Study III

A newly designed booster adapter section of the Agena vehicle was subjected to a constant temperature test of 200°F for four hour periods (Figure 14).

During the temperature soak strain gauges previously located on the vehicle provided information which enabled the design and fabrication engineers to verify their design integrity.

In addition, transducers which will provide inputs to the environmental telemetry during flight were calibrated, thereby insuring the acquisition of accurate flight information. The most economical way of obtaining this environment was to encase the booster adapter section completely with a uniform heat flux as obtained using the HIVOS heat flux simulator.

CONCLUSIONS

The space simulator of today can provide dependable long duration testing at low pressures with accurately controlled heat flux programming for thermal simulation. These facilities provide the design, integrity, and reliability testing information urgently needed for the space vehicles of today and tomorrow. Unfortunately, the schedules and budgets of some space systems do not permit sufficient testing to be performed so that design discrepancies can be uncovered and modifications incorporated prior to launch. It is hoped that perhaps this article may stimulate the engineers and scientists of the future to incorporate the philosophy of simulated environmental testing and provide sufficient time to incorporate adequate testing.

The space simulator should also be considered as a research and development tool to perform special tests associated with the performance of materials in space.

The design of space chambers of tomorrow will require additional operation parameters--larger size, capable of testing vehicles at least 24 feet in diameter and 50 feet high; high vacuum; safe accessibility for human entrance; solar simulation; rapid pump-down to simulate vehicle ascent; vibration for engine environment, etc.

Future simulators may require double-wall design and ion pumps (such as the new 50,000 cubic ft/sec Ultec Pump).

The challenge to the designer is tremendous. For example, where human entry is required, the chamber must be so designed that in case of an emergency a man can be removed from the chamber and placed in an interlock where medical attendants can provide assistance, within 60 seconds. Since all doors must be vacuum tight, new designs and techniques need to be developed. At present a solar simulator of proper size, uniformity and intensity is beyond the existing state-of-the-art. This challenge is one of many which has yet to be overcome.

APPENDIX

Mathematical Derivations

The equation governing the instantaneous heat balance on an element of vehicle skin in orbit can be written as:

$$Wc_p \frac{dT_s}{dt} = F_s S \alpha_s + F_R R \alpha_R + F_E E \alpha_E - \sigma\epsilon T^4 + P_t + Q_{cond} + Q_{cross} \qquad (1)$$

where

W is the weight of the element,

c_p is specific heat,

$\frac{dT}{dt}$ is the rate of the temperature change,

$F_s S \alpha_s$ is absorbed insolation,

$F_R R \alpha_R$ is the absorbed earth reflection,

$F_E E \alpha_E$ is the absorbed earth emission,

$\sigma\epsilon T^4$ is the radiation emitted in space,

P_t represents heat power generated by the internal equipment,

Q_{cond} is the heat being conducted along the wall to the element under consideration,

Q_{cross} is the heat radiation from other internal members to the element.

In the Orbital Simulator, the heat balance equation on the same element can be written as:

$$W c_p \frac{dT}{dt} = \sigma \mathcal{F} (T_w^4 - T^4) + P_t + Q_{cond} + Q_{cross} \tag{2}$$

where

T_w is the test specimen wall temperature,

$\mathcal{F}$ is the geometrical factor between the chamber wall and the specimen skin element under consideration.

The remaining terms are identical to those in Equation (1). Exact temperature simulation requires that at any time the solution of Equation (2) is identical with that of Equation (1); i.e.,

$$\sigma \mathcal{F} (T_w^4 - T^4) + P_t + Q_{cond} + Q_{cross} = F_s S \alpha_s + F_R R \alpha_R + F_E E \alpha_E$$

$$- \sigma \epsilon T^4 + P_t + Q_{cond} + Q_{cross} . \tag{3}$$

For a thermal test model identical to the flight vehicle, the terms defining the internal heat transfer are identical in both cases. Thus,

$$\sigma \mathcal{F} (T_w^4 - T^4) = F_s S \alpha_s + F_R R \alpha_R + F_E E \alpha_E - \sigma \epsilon T^4 . \tag{4}$$

If Equation (4) can be satistifed, the internal and external temperature history will be identical to the test flight. It might be pointed out that factor can be made equal to the emissivity . Factor is given by

$$\mathcal{F} = \frac{1}{\frac{1}{\epsilon} + \frac{1}{\epsilon_w} - 1} . \tag{5}$$

For concentric surfaces, if we make ϵ_w equal to 1 (black body), then the Equation (5) will reduce to

$$\mathcal{F} = \epsilon .$$

Therefore, Equation (4) can be written

$$\sigma \epsilon T_w^4 = F_s S \alpha_s + F_R R \alpha_R + F_E E \alpha_E . \tag{6}$$

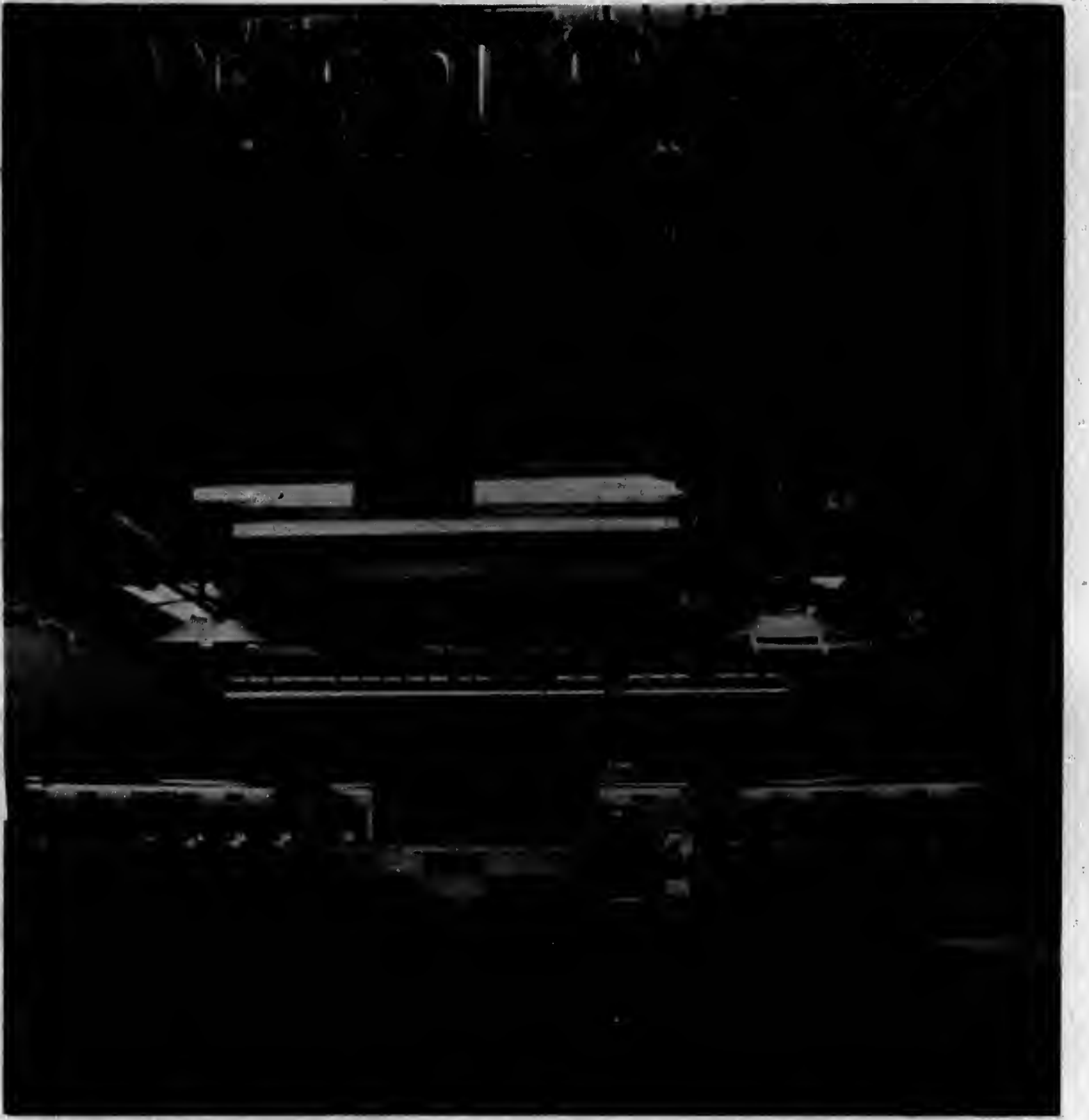


Figure 1. The Lockheed High Altitude and Temperature Simulator

[redacted]

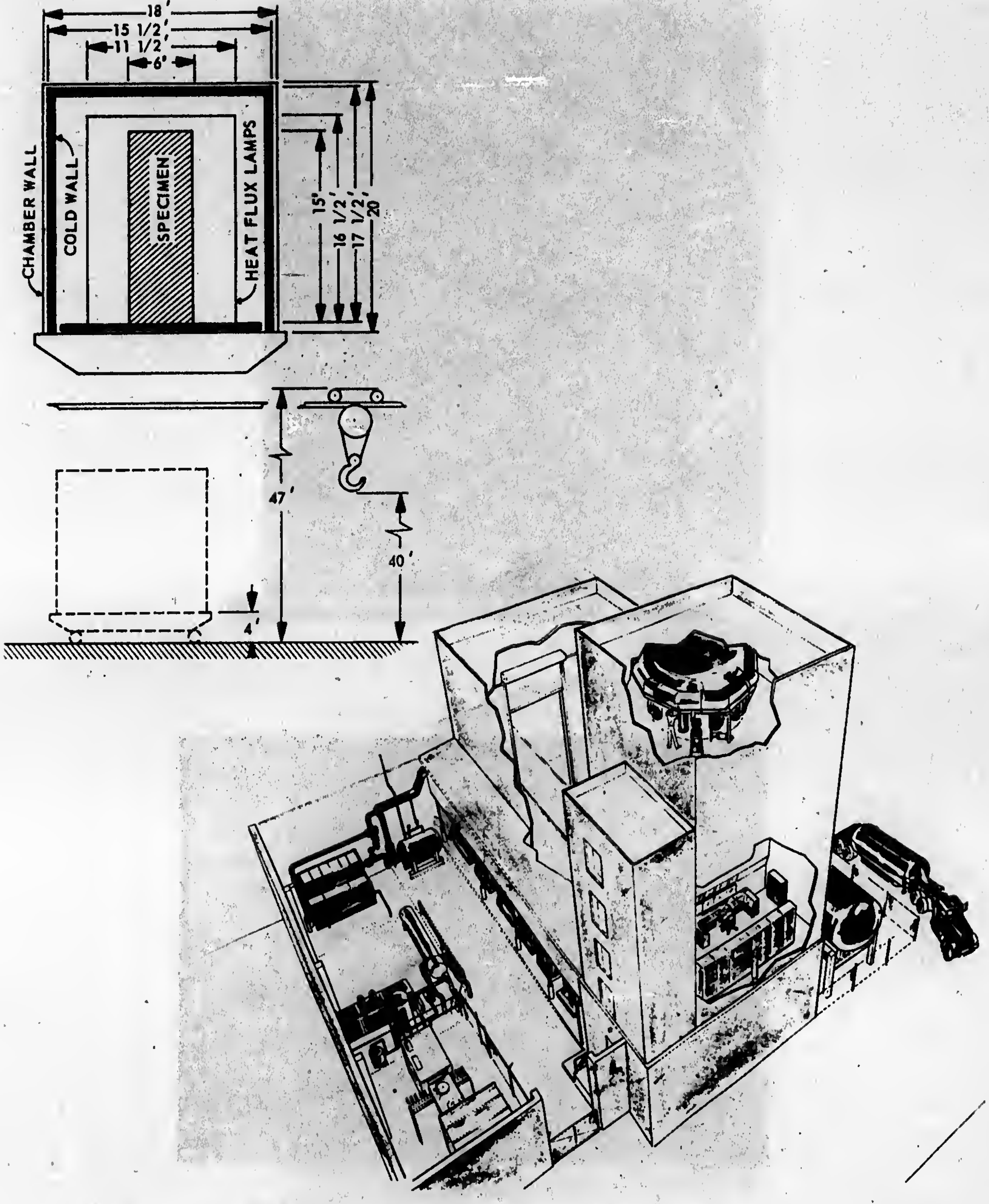


Figure 2. The Lockheed Missiles and Space Division High Vacuum Orbital Simulator.

[illegible]

[illegible]

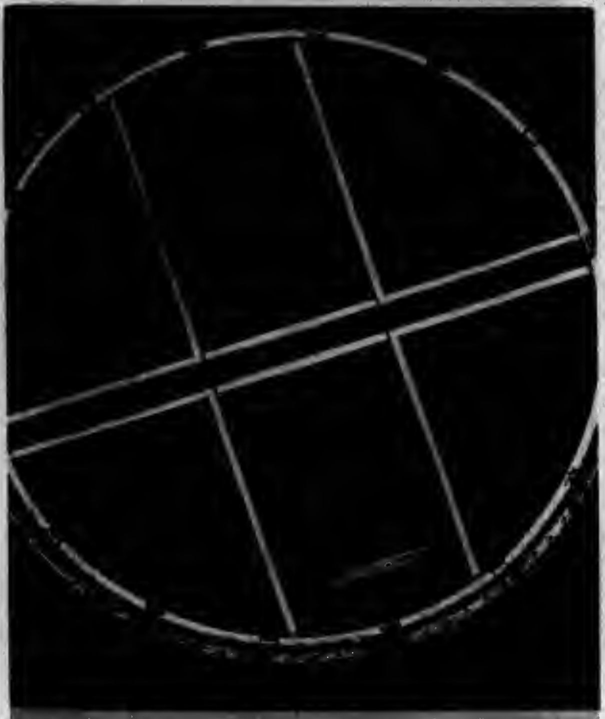


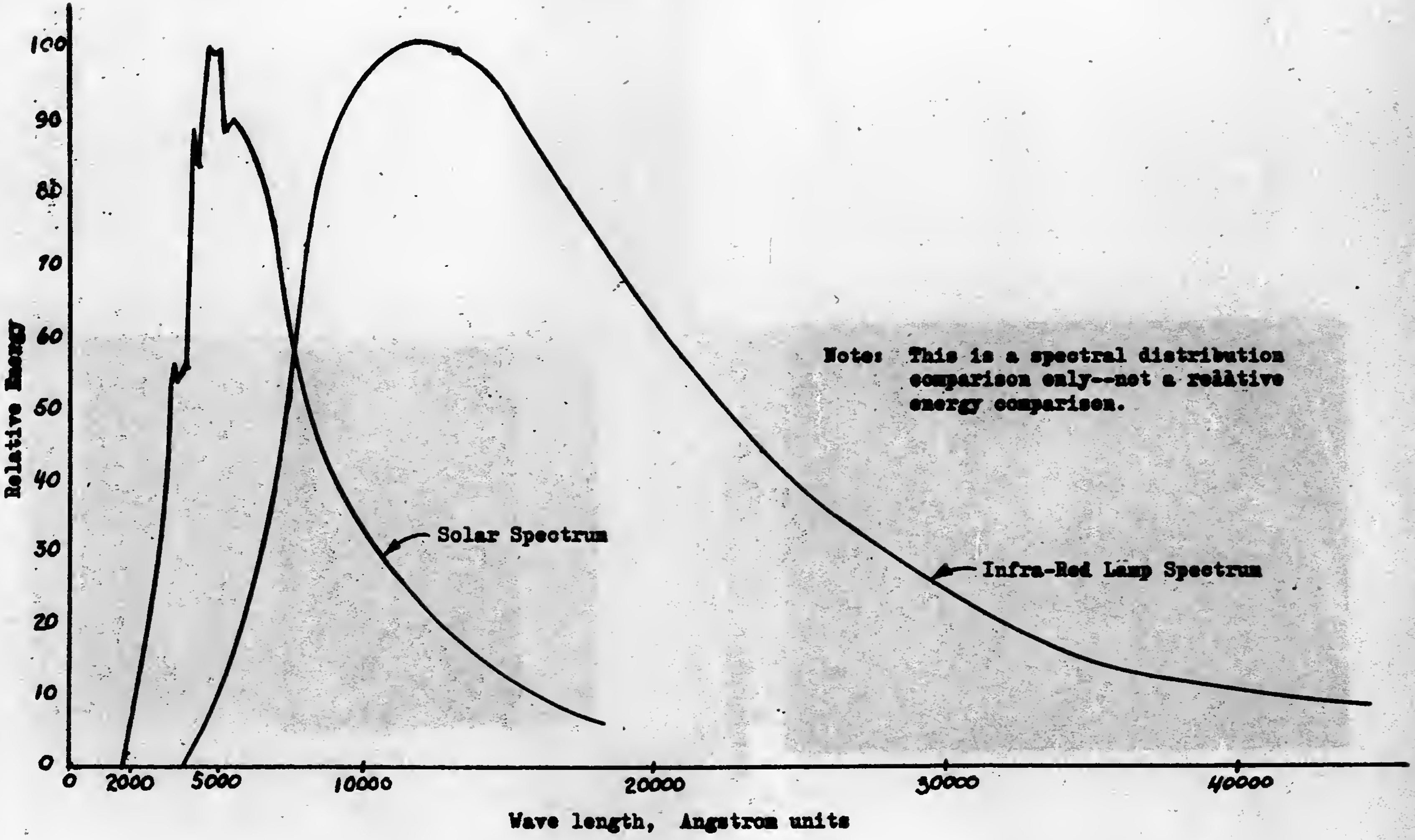
Figure 5. A View of the KIVOS Chamber from the Base looking overhead.

Figure 7. Base of the Instrumented End Bell showing the Lower Sections of the Heat Flux Simulator and the Chevron Plates of the Cold-wall.

Figure 8. The Heat Flux Simulator being readied for final checkout prior to testing.

[illegible]

Figure 10. The Automatic Control Panel which monitors and programs the vacuum system, liquid nitrogen system, and the Low Temp bake-out system. Vehicle being readied for test is visible in the background.



Note: This is a spectral distribution comparison only--not a relative energy comparison.

Figure 11. Comparison between Spectral Distribution of Solar Radiation from T-3 Quartz Infra-red Lamps in HIVOS Simulator.

[illegible]

13. [illegible]

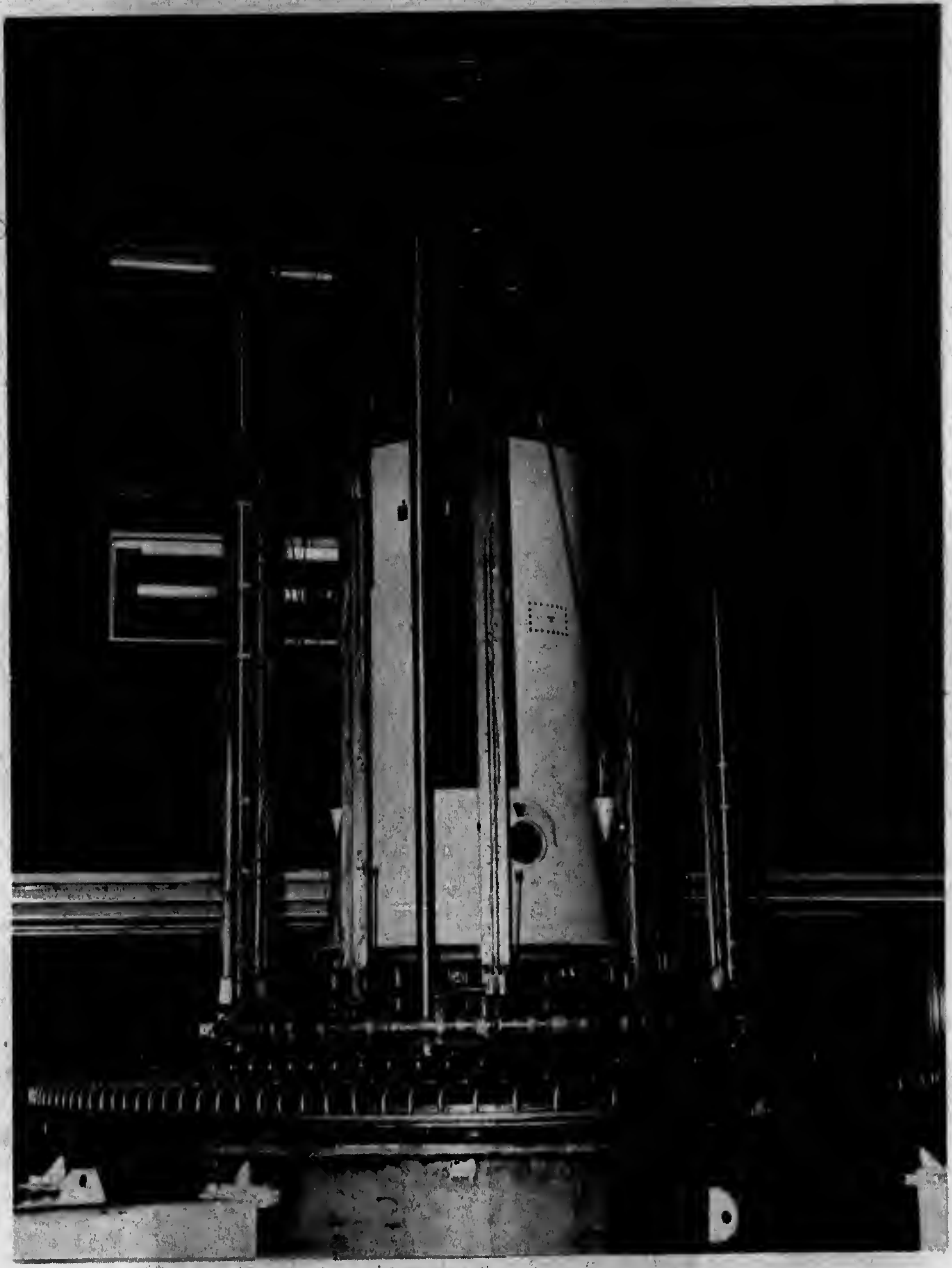


Figure 14. Agena Booster Adapter mounted on the HIVOS Chamber end bell prior to thermal calibration test.

PRECISE TRACKING THROUGH THE UNCERTAIN ATMOSPHERE

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Summary

Bias errors caused by tropospheric refraction may now be corrected to a level below the threshold of the best radars. In most cases, only surface weather data is needed for this. The errors which remain will fluctuate at frequencies from one cps down to one cycle per day, or less, and will lead to serious errors in both radar and interferometer systems. Long baselines reduce the error, but the presence of long-period components in both space and time causes the error to drop very slowly as the baseline is extended. Effects of these components on position and velocity data are shown. Although ionospheric errors can be held to the required level by using frequencies above 3000 mc, it is concluded that precise tracking systems must use trilateration over very long baselines to achieve the accuracies now desired.

Introduction

Instruments available for tracking of aircraft and missiles during the 1940's were limited in accuracy primarily by internal defects rather than by the atmosphere. As development proceeded toward instruments of greater accuracy, the problems of atmospheric refraction were given intensive study, and procedures were devised for applying partial corrections for range and angle errors introduced by the troposphere. Progress in radio meteorology was rapid enough to keep the residual errors of the troposphere below the level of instrumental error until the late 1950's, when several types of microwave tracking instrument reached the limits of atmospheric uncertainty. Throughout this period, electronic systems operating in or below the VHF band were encountering the ionospheric limits to accuracy which still prevent the exploitation of this portion of the spectrum for precise tracking of high-altitude targets.

We will review here the ways in which the atmosphere introduces error in various tracking systems, and some of the important studies which have provided partial corrections for these errors. Tropospheric problems will be taken first, following which we will make a brief survey of ionospheric effects before summarizing the influence of the entire atmosphere on tracking systems.

Tropospheric Range and Angle Bias

The average refractive effects of a spherically stratified atmosphere on radio tracking instruments have been computed and presented in several ways, using slightly different approximations of the "standard atmosphere".[1-4] To illustrate the magnitude of effects to be expected, we have plotted in Figures 1 and 2 the bias errors in range and elevation angle for the CRPL Exponential Reference Atmosphere which closely matches the world-wide average for sea level stations.[3] The errors are dependent upon target range and altitude, running from several feet to about 300 feet in range, and from several tenths of one milliradian to several mr in elevation angle. Even early tracking radars such as the SCR584 required corrections in order to stay within their error specifications at long range. For aircraft tracking, corrections were applied in range by setting the basic ranging oscillator to a frequency somewhat below that which would have corresponded to the vacuum velocity of light. Computations made on the basis of an assumed earth's radius equal to 4/3 times its actual value could be used to remove some of the error in elevation data. If greater accuracy were required, the instrumentation user could select from the literature an atmospheric model and a correction procedure which met his average site conditions, target flight range, and accuracy requirements, and expect to reduce the "bias" errors to a point below the measurement threshold of his system. For tracking with radars derived from the SCR584, this level was about 20 feet or 0.3 mil rms, thus calling for removal of, at most, 90% of the initial error.

Correction of Bias Variations

Bias error corrections based on the standard or local average atmosphere were not of sufficient accuracy to meet the needs of the 1950's. Variations in refractive index profiles, from place to place and over periods of hours to months, caused substantial changes in range and elevation refractive error, and also introduced some error in azimuth measurements. Two general approaches were used to reduce these errors: measurement of spatial and temporal variations in refractivity to improve the correction procedures, and development of new instrumentation systems which exhibited reduced sensitivity to the fluctuations.

The first step in refinement of refraction corrections involved the use of surface measurements to adjust the scale of the refractivity profile. Considerable success was achieved in predicting the total atmospheric bending and range error (as would apply to targets well above the atmosphere), using measurements taken on the ground.[5,6] This procedure was also extended to targets within the atmosphere, leading to correction procedures which were adequate for use with a radar of the AN/FPS-16 class over most of its operating region.[7,8] For instance, the following simple formulae gave results which permitted the achievement of system accuracies near 10 feet rms in range and 0.1 mil rms in angle for targets above 6°:

$$\Delta R_E = 0.0335 \frac{N_s}{\sin E_o} \times \frac{h + 0.1\, h^2}{50 + h + 0.1\, h^2} \qquad (1)$$

$$\Delta E_o = \frac{N_s}{\tan E_o} \times \frac{h}{35 + h} \qquad (2)$$

Here, ΔR_E is the range error in feet, N_s the surface refractivity in N-units (or parts per million of the refractive index), E_o is the elevation angle, h is the target altitude in kilofeet, and ΔE_o is the elevation error in microradians.

Although the accuracy could be improved further by using more complex computation procedures, significant gain in accuracy for the case of targets within the atmosphere is dependent upon use of actual measured profiles of refractivity. An estimate of the optimum correction accuracy for radar data, based on the average of several expert (and conflicting) viewpoints, was given by a committee of the National Academy of Sciences last year, and is shown in Table I below.[9]

Sensitivity of Instrumentation Systems to Refraction

Development of instruments which would have reduced sensitivity to atmospheric (as well as instrumental) sources of error was carried out during the 1950's, leading to the so-called baseline type of tracking system. From the standpoint of refraction error, we obtain two advantages by performing measurements over a long, horizontal baseline. It can be shown that the first-order elevation bias error, in the long-range case, is absent from this system.[9,10] It is therefore unnecessary to measure N_s in order to correct the elevation bias error at high elevation angles, since the term $(N_s/\tan E_o)$ from Eq. (2) is applied by the geometry of the instrument. A second advantage lies in the ability of long-baseline systems to reduce the high-frequency components of fluctuating error, as shown in Figure 3.

This figure emphasizes the common basis for measurement of target position in all radio tracking systems: the comparison of phase or time delay at different points on the received wave front. The pulse radar, Figure 3a, measures the round-trip time delay of the envelope of the transmission to obtain range, and adjusts the plane of its antenna so that it lies parallel to the phase front of the received r-f signal, averaged over the antenna aperture. (The pulse Doppler radar also uses the r-f phase information to refine its range precision). The interferometer, Figure 3b, samples the received phase fronts at two or more separate points, and computes the angle of arrival from knowledge of wavelength and system geometry. Ambiguities in the measurement are resolved by using multiple antennas with different spacing. The effect of phase fluctuations which produce one or more cycles of "ripple" in the wave front between the antennas is reduced, as compared to the smaller tracking antenna, although a continuous antenna of equal length would do an even more effective smoothing job if it could be used. Phase fluctuations whose wavelength is longer than the interferometer baseline will produce the same error in either case.

A third type of system, Figure 3c, extends the baseline beyond the point where r-f phase ambiguities can be resolved, and locates the target by "trilateration", comparing the envelope delay (or phase difference of the modulation waveform) as observed at the two sites. From the point of view of tropospheric refraction, it makes no difference whether the carrier or the envelope delay is measured, and the significant difference between systems is the maximum extent of the baseline and the number of samples (if any) which are gathered between the end points. Angle-tracking antennas are limited to about 200 feet in diameter, for economic reasons, and generally are in the 10 to 100 foot class. Interferometers have been built to use spacings of 100 to 100,000 feet, although resolution of phase ambiguities is not generally possible beyond a few thousand feet in microwave systems. Trilateration systems may be extended to approach the diameter of the earth, and further extension to satellites is not impossible if earth-based systems prove inadequate.

When the interferometer systems were first developed, it was believed that fluctuations in tropospheric refraction were, for the most part, limited in size, and that extension of baseline length beyond this size would result in errors almost inversely proportional to baseline length.[11] According to this theory, an Azusa system using a baseline of 150 feet would be at least ten times as accurate as a radar with a 15-foot antenna. A Mistram, using a baseline of 10,000 feet for position measurements, would be better than Azusa by a further factor of sixty, while its velocity data (from a baseline of 100,000 feet) would be 600 times better. Even after allowing for the foreshortening of the effective baseline (GDOP) for elevation measurements on targets removed from zenith, these improvement factors would have justified considerable effort and expense in an interferometer system. The reasons for the failure of interferometer systems to meet these expectations have been obscured by difficulties in establishing standards against which the errors can be measured, but the results of atmospheric studies provide clear evidence that the initial assumption as to the size of the tropospheric fluctuations was in error.

The Nature of Tropospheric Fluctuations

Refraction studies performed at the Central Radio Propagation Laboratory of the National Bureau of Standards had provided by early 1960 a description of the frequency spectrum of tropospheric fluctuations.[12] A summary of this data is shown in Figure 4, where the power spectral densities of range and refractivity variations for two paths are plotted on logarithmic scales covering several decades. The range fluctuation is in terms of parts per million of the total path length, which was between 10 and 15 miles at an elevation angle of about 6°. The most notable feature of the range spectrum is its steady increase in intensity with decreasing frequency, down to one cycle per day, and the absence of any overwhelming daily component. This suggests that the refractivity variations are not the result of changing temperature, but may reflect the drift of a more or less "rigid" pattern of anomalies across the measurement paths.[13] Later measurements of phase difference variations across baselines of several lengths tended to confirm the equivalence of the temporal and spatial spectra.[14] The top scale of Figure 4 has been added by this writer to the NBS plot, and used to derive the errors which would have been observed across the Hawaii baselines if the fluctuations had resulted entirely from atmospheric drift at 10 ft/sec. The results are compared in Figure 5 with the measured data, expressed in terms of angle error for different baseline lengths (for the 15-mi path at 6°).

The Hawaii baseline data was taken over limited periods of time, and hence could not show the low-frequency components of error which are predicted by the spectrum of Figure 4. In addition, since the study was intended to measure variations only above about 0.003 cps, special efforts were made to eliminate the longer-term "trends" in the data, by selection of samples and by fitting the data to smooth curves. These low-frequency components, however, are the errors which would produce the largest effects on long-baseline systems, and their inclusion leads to the solid curve on Figure 5. If they are removed by a filtering process from the spectrum of Figure 4, the calculated errors agree almost exactly with measured data from which "trends" were removed. The conclusion which has been drawn from the NBS baseline data[14], that errors will vary inversely with baseline length for baselines beyond about 3000 feet, is only true if we are willing to ignore the error spectrum below 0.003 cps. If these low-frequency errors are important to us (and we will show later that they are of great importance in tracking targets with appreciable angular rates), then we must consider that the error varies only as the inverse fourth root of baseline, for baselines less than 100,000 feet. When compared with antennas whose aperture is large (e.g. 30 to 100 feet), the reduction in error with baseline length will be negligible until this length reaches several hundred feet.

The spectrum of Figure 4 also provides a means of predicting the predominating frequencies of refractive error in baseline systems of any length. When we combine the fluctuation spectrum W(f) with a filter function $G_b(f)$ representing the response of the baseline, as in Figure 6, we obtain the spectrum of the range-difference error across that baseline. The variance of the error component in a given octave or decade of frequency is proportional to the spectral density times the center frequency of the octave or decade.[15] Thus, the largest error component will appear near the frequency $f_b = 0.22\ b/v_w$, where b is baseline length and v_w is drift velocity across the measurement path. Substantial error components will spread over the frequency range from one-tenth to ten times this frequency. For the assumed 10 ft/sec wind velocity normal to the measurement path, the largest component will appear near 0.03 cps for b = 100 ft, 0.003 cps for b = 1000 ft, and 0.0003 cps for b = 10,000 ft. If the low-frequency components are removed by limiting the observation time t_1 or by intentional high-pass filtering, the error will appear largest at the low-frequency cut-off of this filter, if this is above f_b. This fact explains the observation of "cyclic" errors in such systems as Azusa.[16] As the observation period is increased, the magnitude of the error and the amplitude of the low-frequency component will grow larger, until all components down to about $f_b/10$ are observed or until the system is recalibrated often enough to suppress the additional low-frequency error. We should note that the low-frequency error components, although very hard to observe in many tests, can lead to significant errors in prediction of missile impacts or satellite orbits. The measured data can appear free from undesired "noise", and yet contain trends which degrade its accuracy in a more serious way (noise can be removed by data smoothing, while low-frequency error cannot). Verification of this effect is found in monthly reports of instrumentation accuracy, which over many periods have shown less than two-to-one advantage in impact prediction accuracy for Azusa over tracking radar, although the advantage in tracking accuracy is represented to be nearer ten to one. The difference lies, presumably, in the fact that much of the radar error is in that region of the spectrum where it may be recognized (and smoothed), while the major interferometer error remains hidden under the actual changes in missile position and velocity. When trajectories measured by different "precise" instruments are compared, the differences in position are frequently far greater than permitted by the confidence limits assigned to the measurements. Although calibration and operating procedure may be partially responsible, much of the spread in data must be attributed to unrecognized atmospheric fluctuation.

Correction of Low-Frequency Fluctuations

The subject of correction for fluctuating errors has been analysed quite thoroughly by Thayer,[17] and we will not repeat his discussion here. We may note, however, that the corrections are effective only in the frequency region below 10^{-4} cps, and hence that they provide significant advantages only for baselines longer than about 100,000 feet. Furthermore, it appears that any refractivity data gathered at stations within about 300,000 feet of each other should be averaged to obtain a single correction applicable to all stations within that region. Corrections for horizontal gradients of refractivity do not appear profitable unless a great deal of additional data is available over a range of altitudes up to about 10,000 feet.

Effects on Velocity Data

When the presence of low-frequency fluctuations became obvious, hopes for extreme positional accuracies in interferometer systems faded. It was still expected, however, that the velocity data specifications might be met, since the derivatives of these slow fluctuations would contribute little to the over-all error. The spectrum of the velocity error for a baseline system is found by multiplying the position error spectrum, Figure 6, by $(2\pi f)^2$. In the case of the 1000-foot system, the resulting spectrum appears as in Figure 7, with most of the error in the band from 0.01 to 1 cps. In this velocity spectrum, it makes little difference whether the very low frequencies are included or not, and the angle velocity error does indeed vary inversely with the baseline length (above $b = 100$ ft), as shown in Figure 8. However, the original fluctuation spectrum, Figure 4, applied only to a fixed path, and the frequencies of the fluctuations depended upon drift velocity of the atmosphere across this path. If we are attempting to measure velocity, we must assume that the measurement paths follow the target, and move through the atmosphere with a velocity which depends upon the angular rate of the target, or its range and tangential velocity. To the extent that the original spectrum could be attributed to drift of a rigid pattern of refractivity across the path, we must now modify its frequency and amplitude scales to account for the relative velocity of moving paths in the atmosphere. The new relative velocity may range from 10 to 100 times the wind drift velocity (in Norton's recent paper[15] he gives the factor as 50 to 500). The position error remains the same, as the increase in frequency scale is matched by a decrease in spectral density. However, the velocity error spectrum contains also the factor $(2\pi f)^2$, and this increases the rms velocity error in direct proportion to the relative velocity. This increase is shown in the upper curves of Figure 8, which apply to a typical satellite track at long range, where the midpoint of the tropospheric path moves at 250 ft/sec. The additional error will not be apparent under controlled test conditions on fixed targets, nor will it appear when a missile is tracked on a path at constant angle, as in some guidance applications. It will be present, however, whenever the paths move through the troposphere at velocities greater than the normal wind drift speed. The new error components will be at higher frequencies, and therefore may be smoothed more effectively in data filters of limited lag time. Curves are shown for smoothing times of two and 20 seconds (lags of one and 10 sec). The effect of smoothing is to equalize the error for all systems whose baseline (or aperture) is less than a certain length. Systems of longer baseline will exhibit errors whose period exceeds the smoothing time, and whose magnitude varies inversely with baseline length.

Ionospheric Errors

The refractivity of the ionosphere is subject to a much greater range of variation than that of the troposphere. In addition to the systematic variation in refractivity as a function of frequency, for a given electron density, there are large variations in electron density with altitude, time of day, time of the sunspot cycle, latitude, and magnetic field conditions. For this reason, attempts to apply corrections for ionospheric errors have met with very limited success. Systems operating at VHF with two well-separated frequencies have been able to obtain corrected data by direct comparison of the two measurements,[18] but the use of ionospheric soundings has provided correction to no better than 50% of the initial error levels, and the same or better accuracy can be achieved by reference to monthly predictions of electron density.[9] Relationships between refractivity, operating frequency, and critical frequency are shown in Figure 9, which indicates the approximate bounds of system operation for precise tracking. Average values of range and angle error for daytime conditions are shown in Figures 10 and 11. Short-term error fluctuations (measured over periods of a few minutes) can be expected to have rms values near 1% of the average error, for distant targets, and perhaps 2-3% for targets within the lower ionosphere. Larger values are to be expected during periods of magnetic disturbance.

Because of the variability of ionospheric error, the preferred solution to precise tracking problems lies in the use of frequencies high enough to bring the average error below the tolerance level. Usually, this implies operation in the band 3000 - 10,000 mc, although tracking to the order of 30 feet is possible as low as 2000 mc. If data is available for correction, it may reduce the error somewhat under "normal" conditions, but the increase in error during disturbances will then lead to a greater spread in system performance levels.

Conclusions

The development of systems and procedures for reduction of atmospheric error has been reviewed, and some of the pitfalls in error estimation have been pointed out. In general, it can be stated that the statistical results of radio meteorological studies have kept pace with the needs of instrumentation system designers and users, but that the demands of the users have gone beyond the level to which the atmospheric errors can be predicted on a deterministic basis. Ignoring some of the important error components caused by long-period tropospheric changes in the measurement path, we have produced instruments which can measure to a precision ten to one hundred times beyond the normal variations in the wavefronts arriving at the antennas, and have neglected the geographical spreading of instrument sites which offers the only solution to precise tracking requirements. In the process, we have learned a great deal about atmospheric variations, but most of this knowledge could have been obtained more economically from experiments of the type performed by the National Bureau of Standards. The problems caused by atmospheric errors is not as bad as might be thought from the previous discussion, because the accuracies of existing instruments are, in fact, adequate for most purposes. Errors which exceed the expectations by factors of ten to one hundred are often not noticed, since there are no standards against which these errors can be measured under actual operating conditions. In the future, however, we can hope that the constraints imposed by the atmosphere on tracking accuracy will be recognized before development is carried out, and that maximum advantage will be taken of those factors which will reduce error in an economical way. Furthermore, we should note that real improvements in system accuracy are unlikely to be made until the actual limits of existing instruments are appreciated by the users. A major step in this direction will have been taken when quality analysis procedures are devised to cover all regions of the error spectrum, rather than the common subdivisions of "noise" and "bias". The mathematical attractiveness of the random-noise plus fixed-bias model is undisputed, but its effectiveness in predicting errors (and impact points) is open to serious question. This is probably the largest single factor producing the "missile accuracy gap" which separates the boundary of the confidence limit of measured data from the actual position or velocity of the target.

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TABLE I

Optimum Accuracy of Range and Angle Corrections

		$\theta_o = 5^\circ$	$\theta_o = 20^\circ$
Long-Range Case ($R > 300$ n. mi.)			
Initial range bias, ΔR_e	(ft)	75	22
Residual range bias, σ_{rb}	(ft)	0.75	0.2
% residual error		1	1
Initial angle bias, δ^t	(μrad)	3500	900
Residual angle bias, $\sigma_{\theta b}$	(μrad)	70	20
% residual error		2	2
Short-Range Case ($R = 50$ n. mi.)			
Initial range bias, ΔR_e	(ft)	22	7
Residual range bias, σ_{rb}	(ft)	0.5	0.15
% residual error		2	2
Initial angle bias, δ^t	(μrad)	2000	700
Residual angle bias, $\sigma_{\theta b}$	(μrad)	60	20
% residual error		3	3

(Values shown should be doubled for disturbed meteorological conditions such as heavy cloud cover, fronts, and inversions; also for lack of reliable and frequent soundings covering the entire tropospheric path used in measurement).

(from Ref. 9)

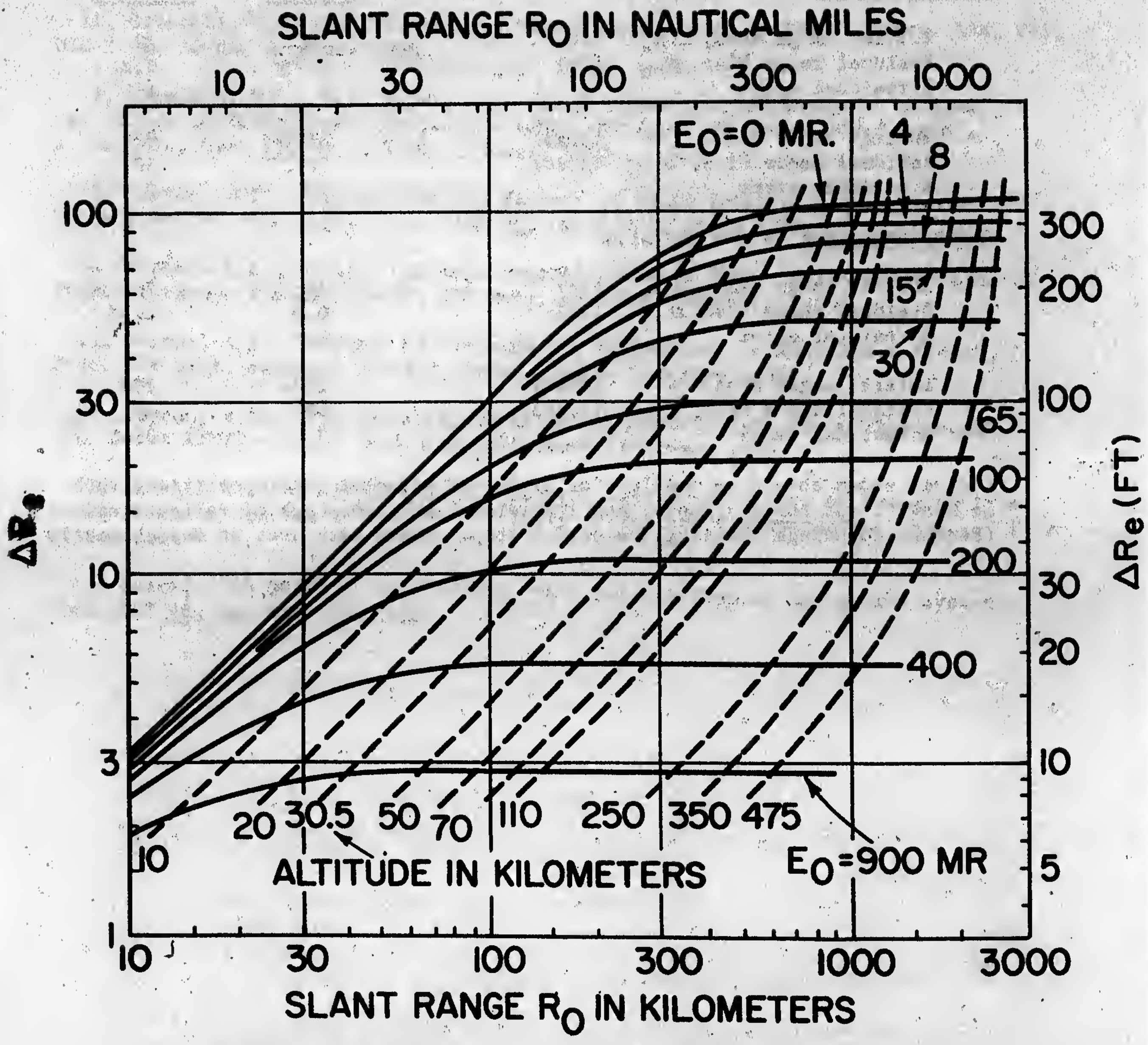


Figure 1. Range bias vs. range for CRPL exponential reference atmosphere, $N_0 = 313\ \mu$ (from Ref. 9).

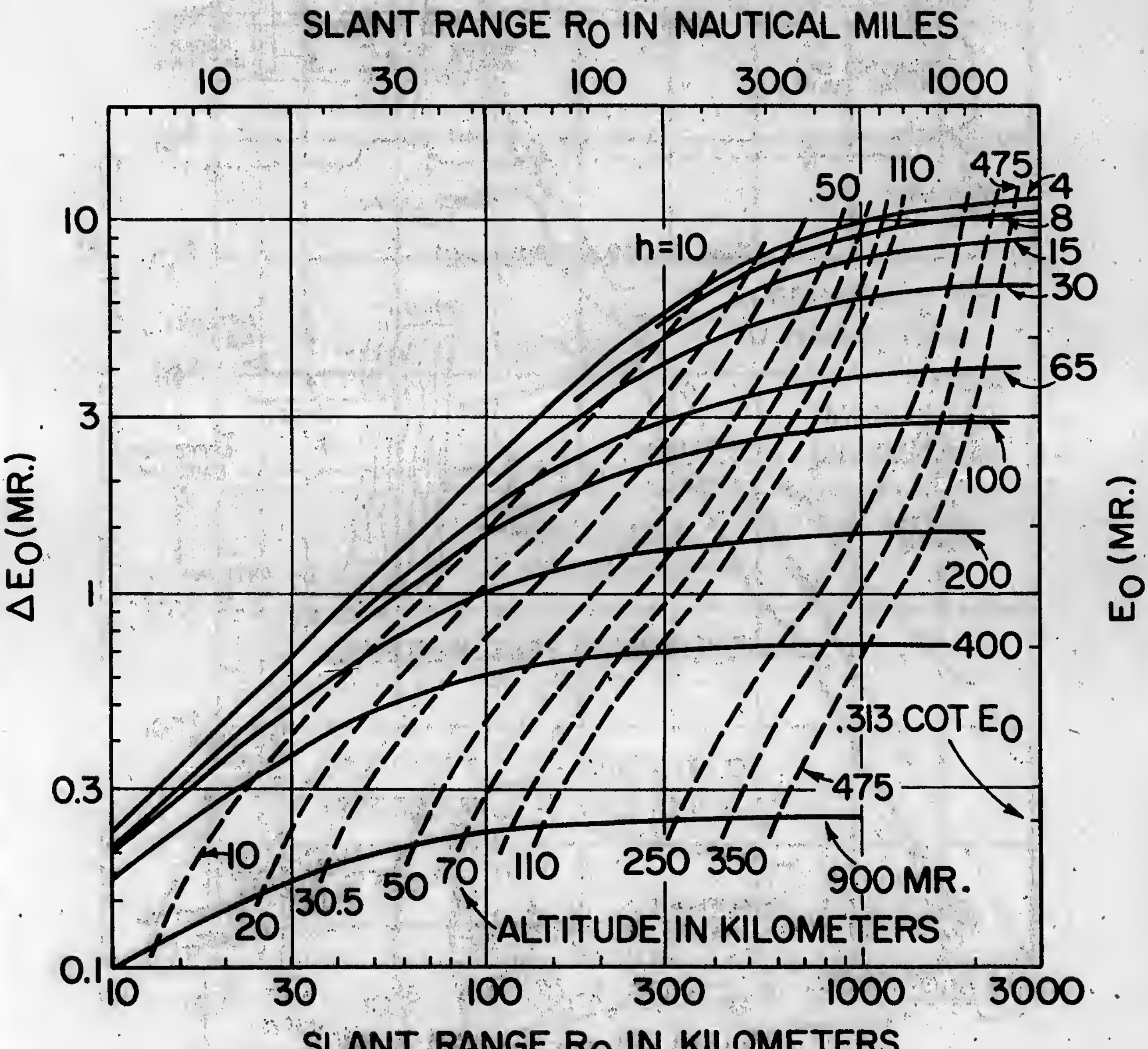
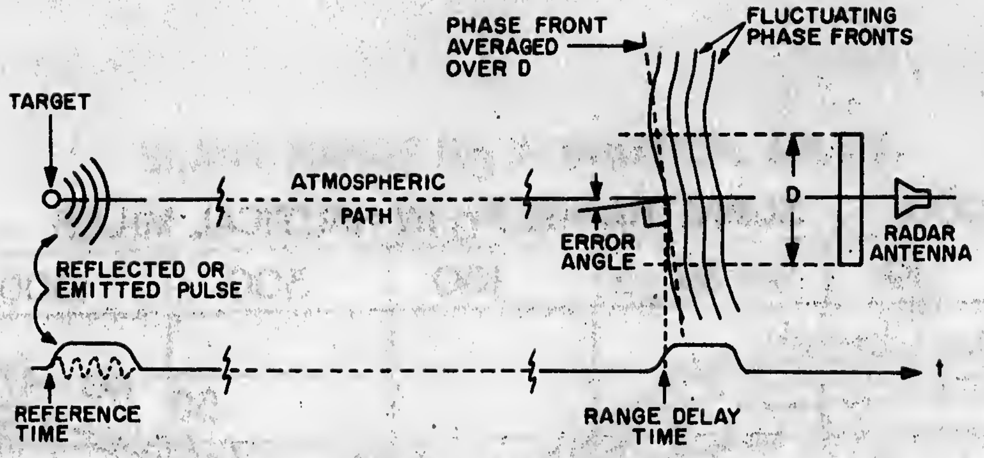
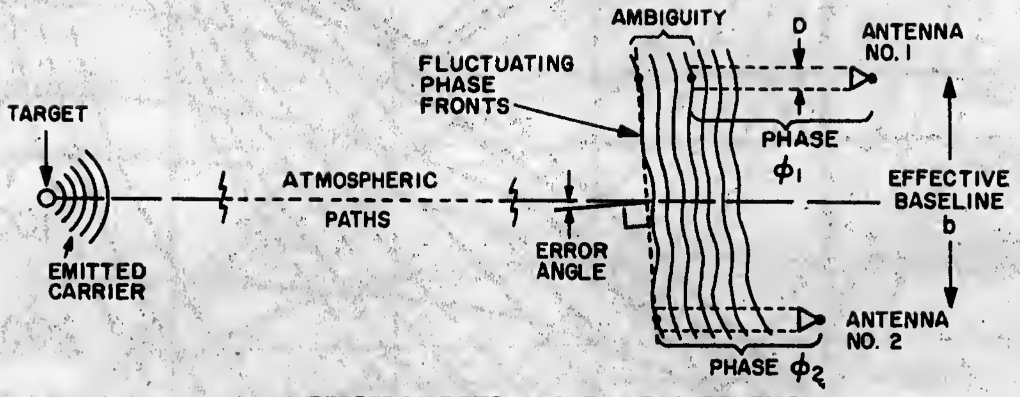


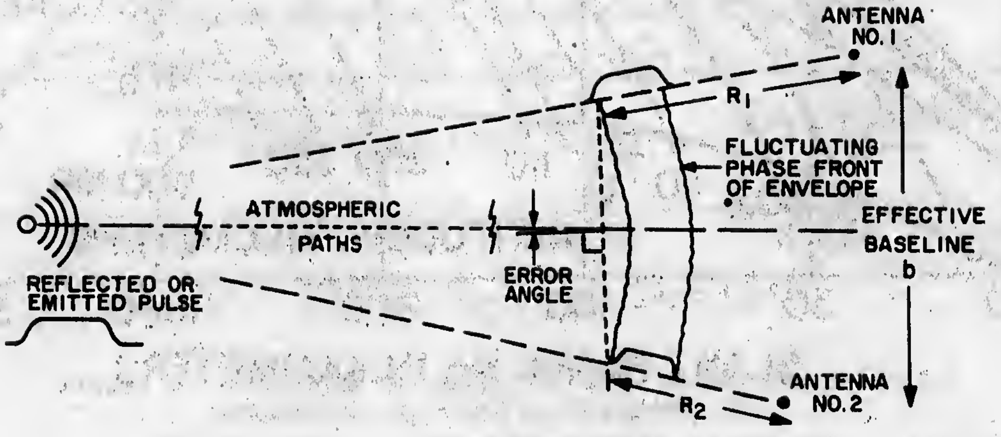
Figure 2. Tracker elevation angle error vs. range for CRPL exponential reference atmosphere, N_o = 313 (from Ref. 9).



a PULSE RADAR MEASUREMENT



b. INTERFEROMETER ANGLE MEASUREMENT



c. TRILATERATION

Figure 3. Fluctuation effects on different instruments.

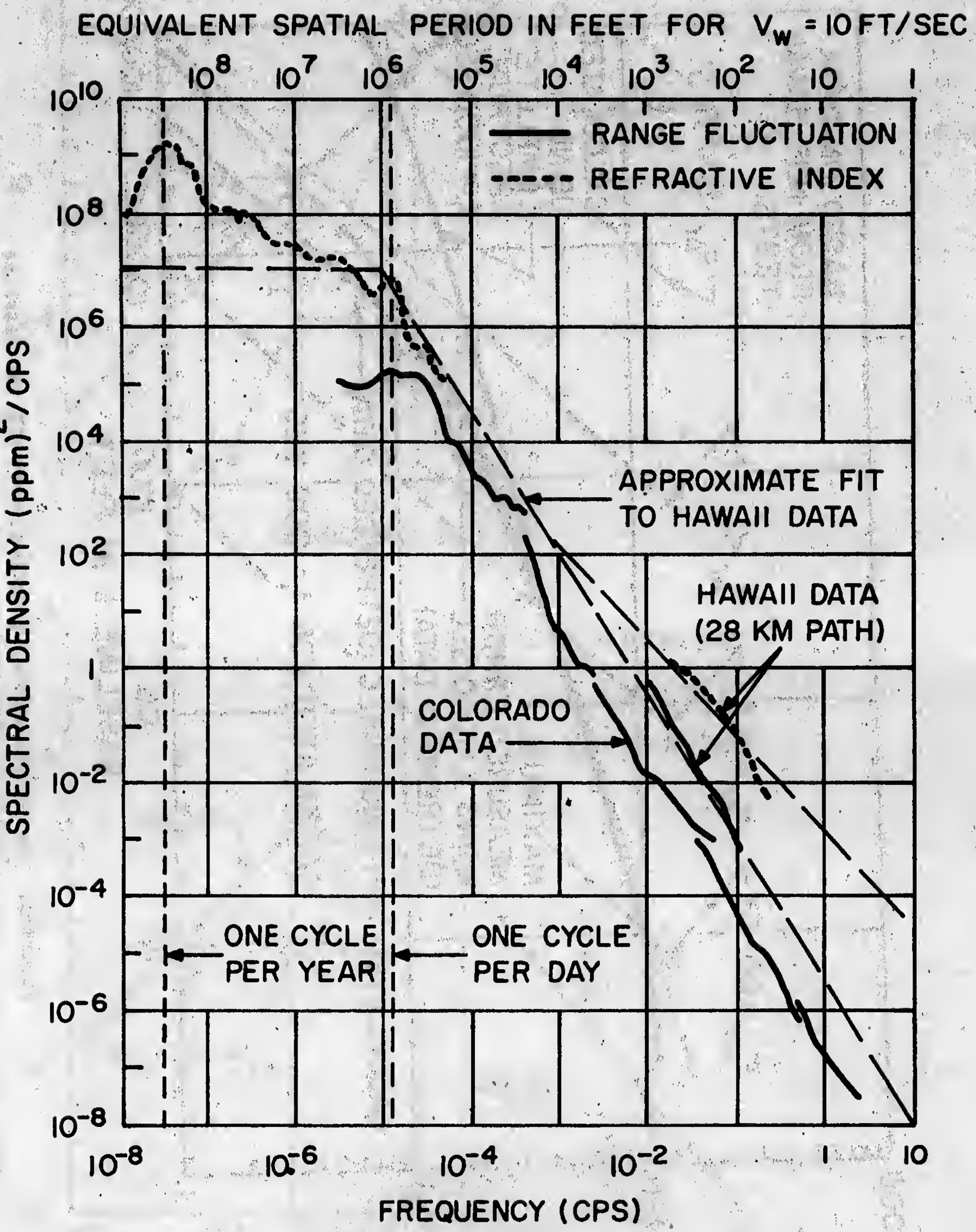


Figure 4. Spectra of refractivity and range fluctuation (NBS data, Ref. 12).

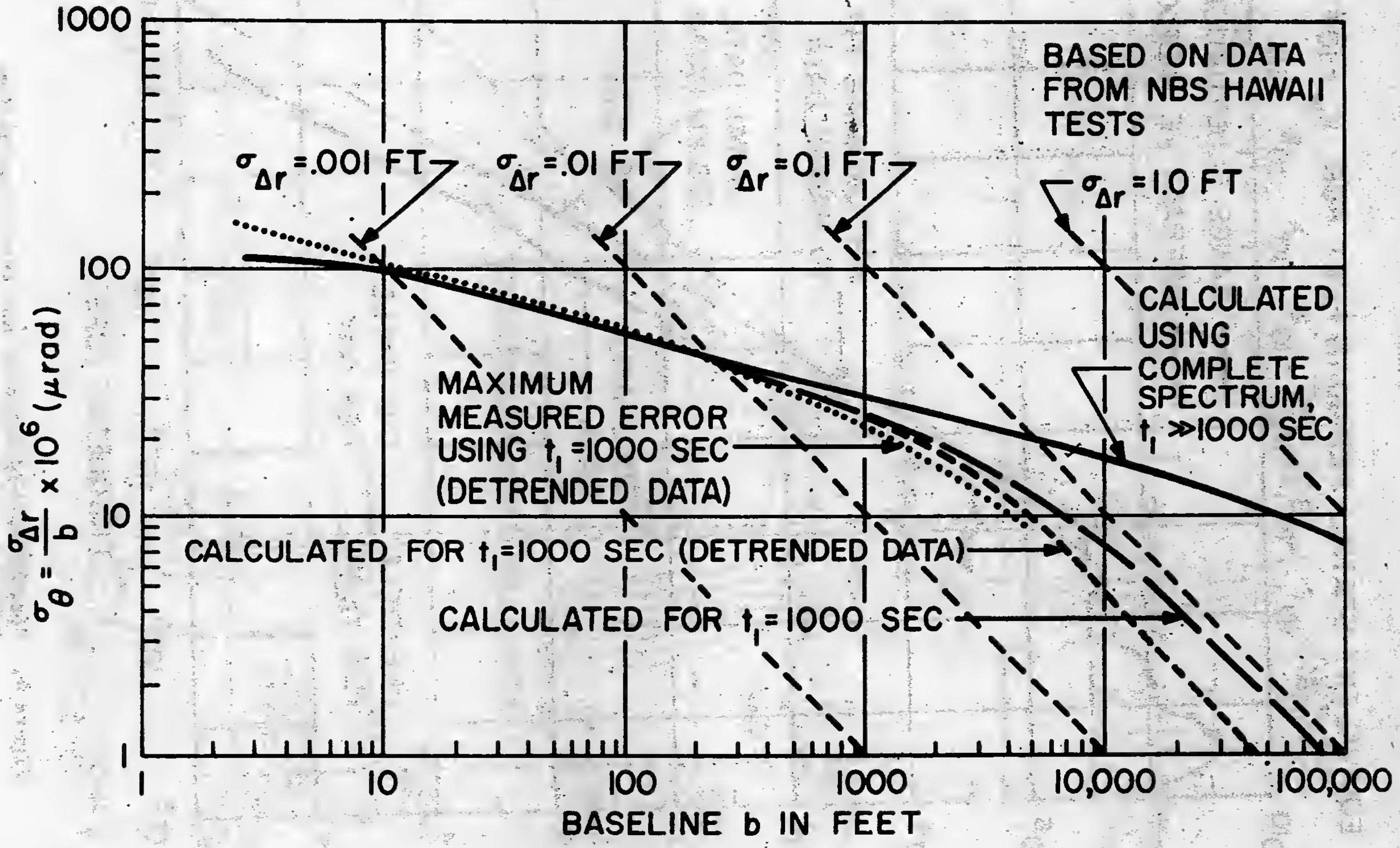


Figure 5. Equivalent angular position error σ_θ vs. baseline length b, for complete spectrum and for limited observation time t_1 (from Ref. 13).

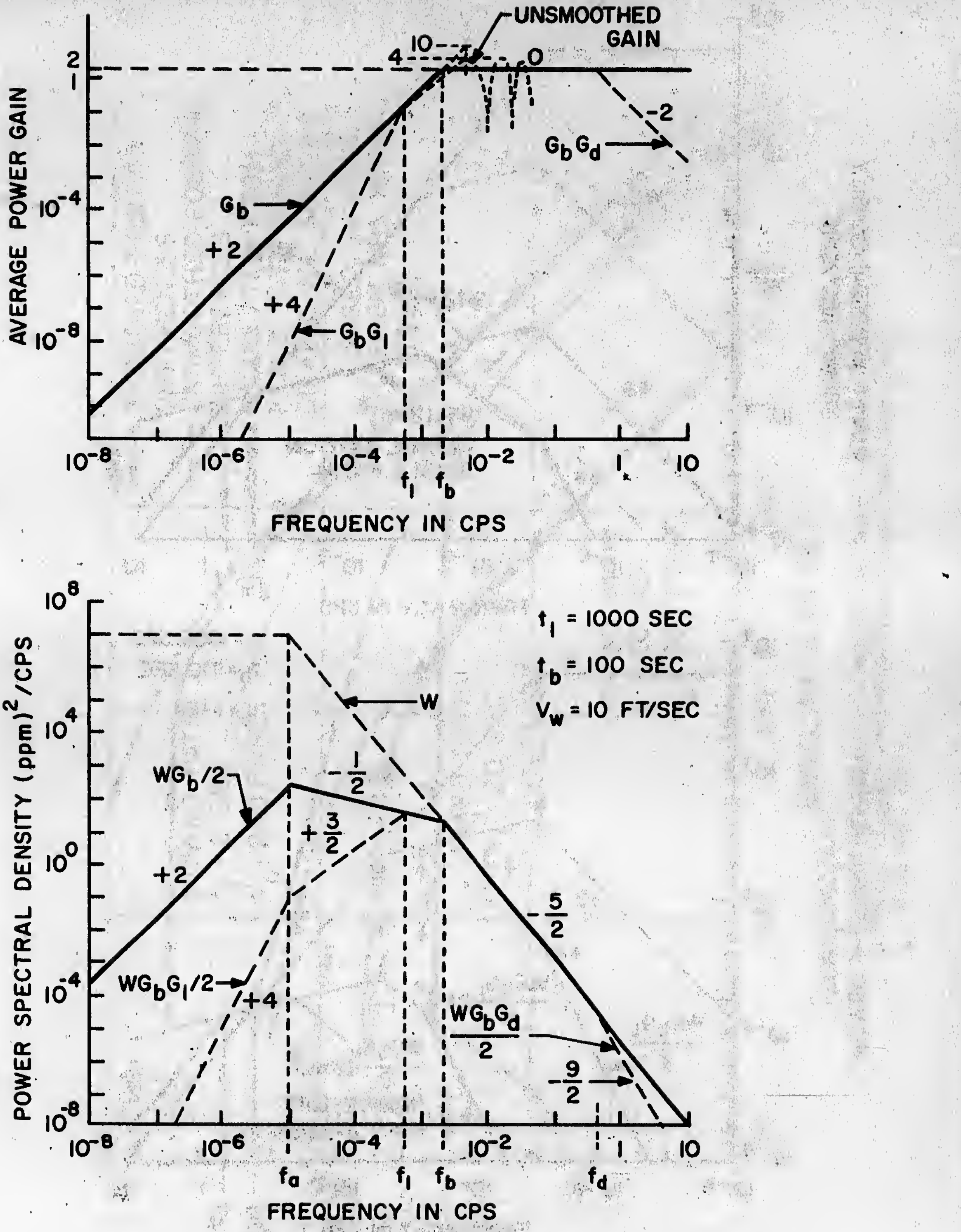


Figure 6. Response function of baseline system, G_b, and resulting range-difference fluctuation spectrum, WG_b, for b = 1000 ft, wind speed 10 ft/sec. (from Ref. 13).

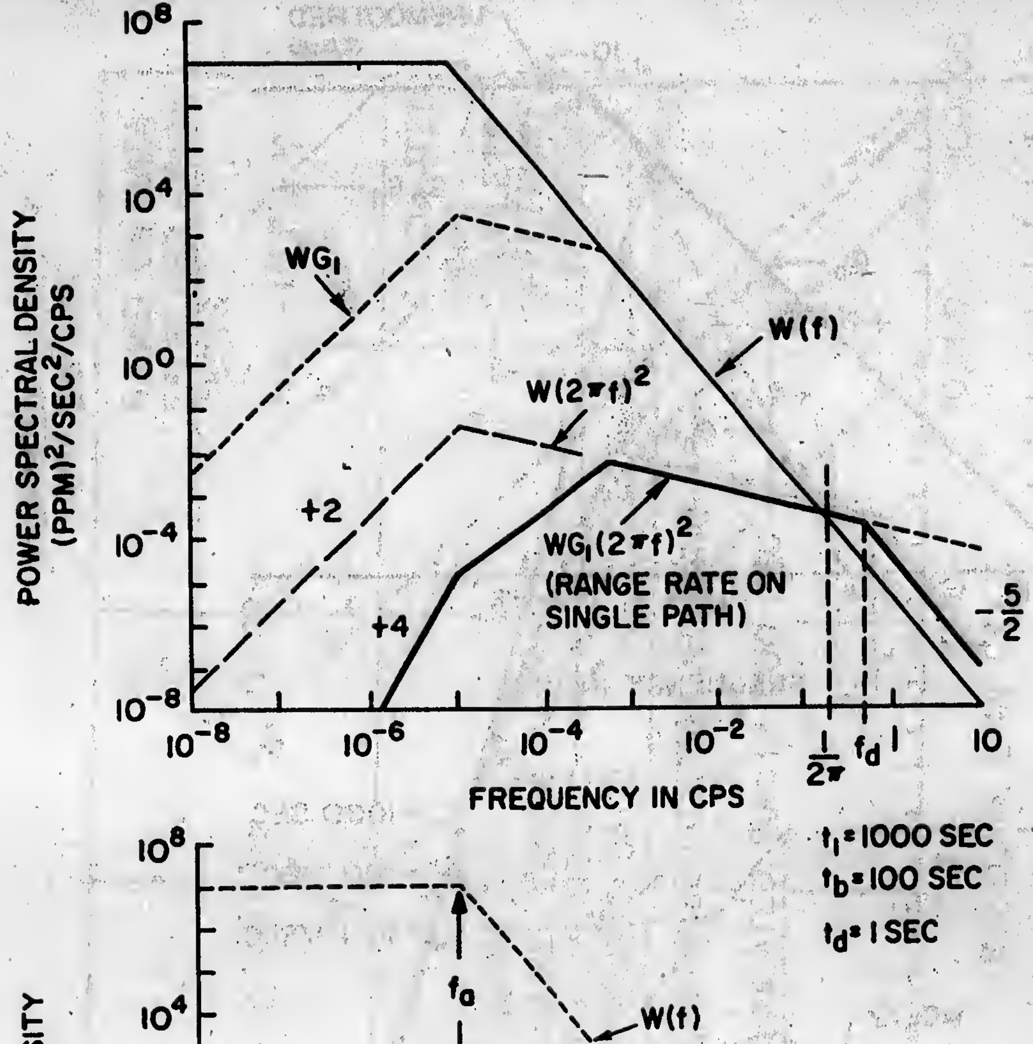


Figure 7.

Velocity and velocity-difference fluctuation spectra, for b = 1000 ft, wind speed 10 ft/sec. (from Ref. 13).

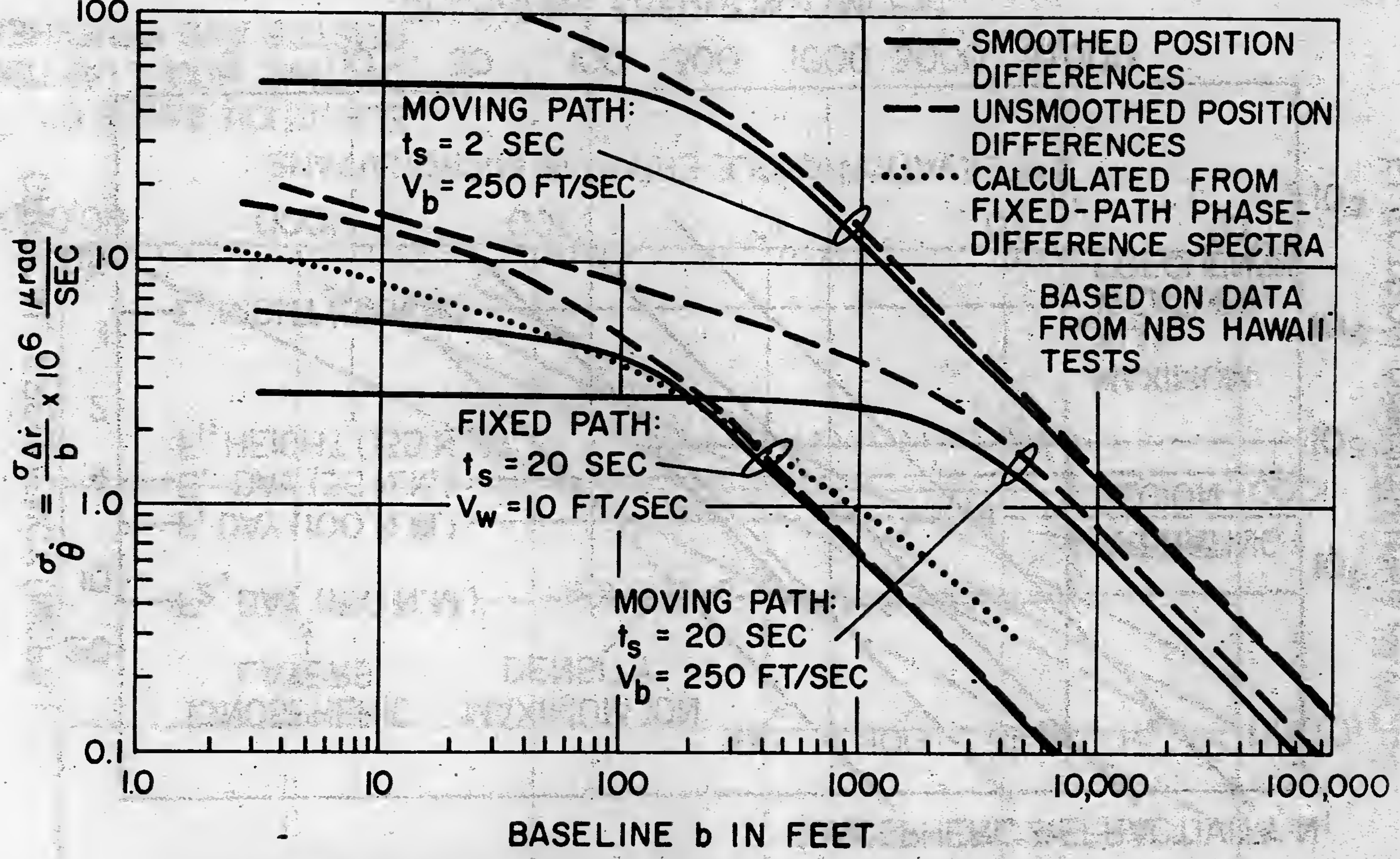


Figure 8. Equivalent angular rate error $\sigma_{\dot{\theta}}$ vs. baseline length b, for different relative beam velocities v_b and smoothing times t_s (from Ref. 13).

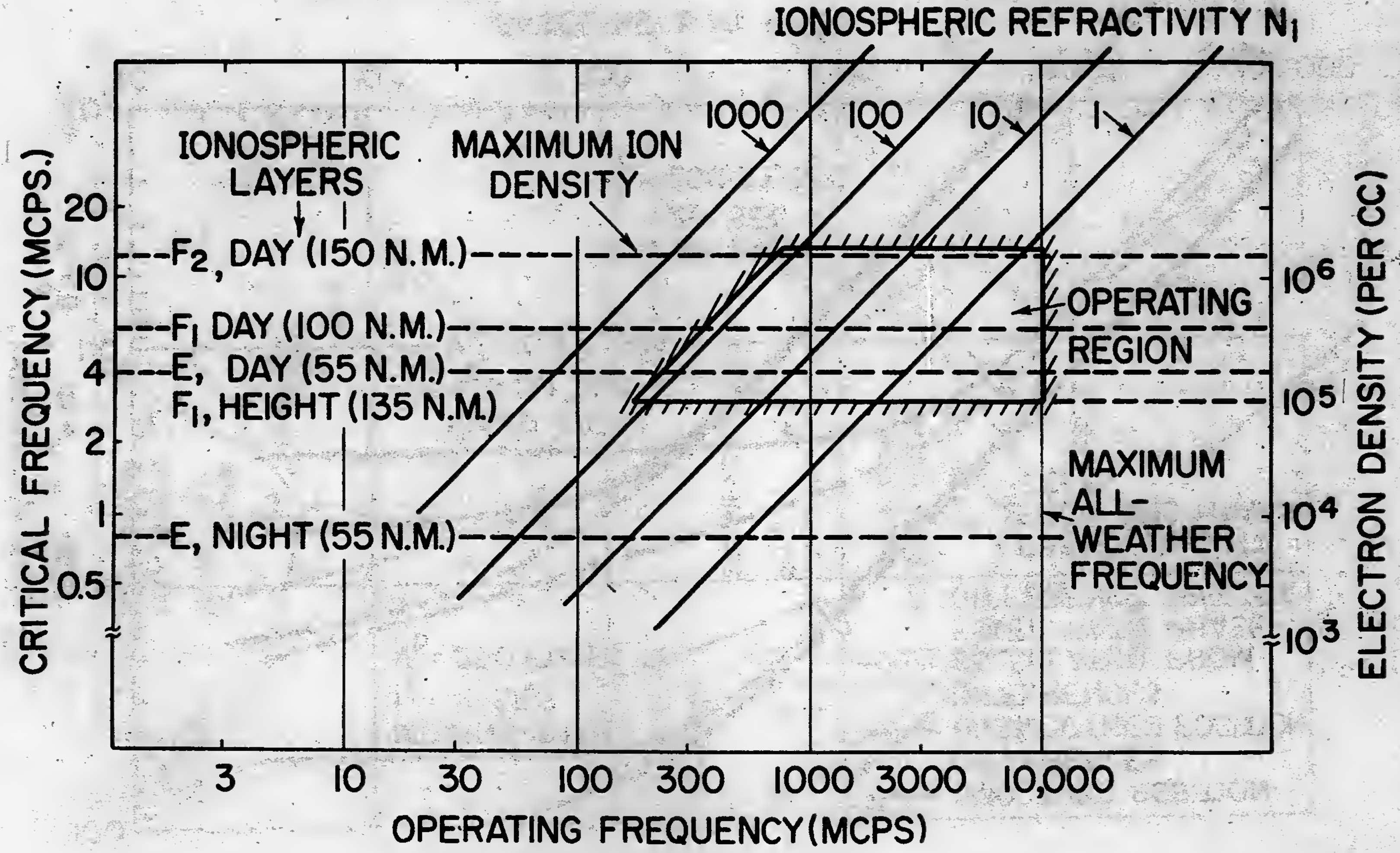


Figure 9. Relationship between ionospheric refractivity N_1, critical frequency, and operating frequency, showing limits for precise instrumentation.

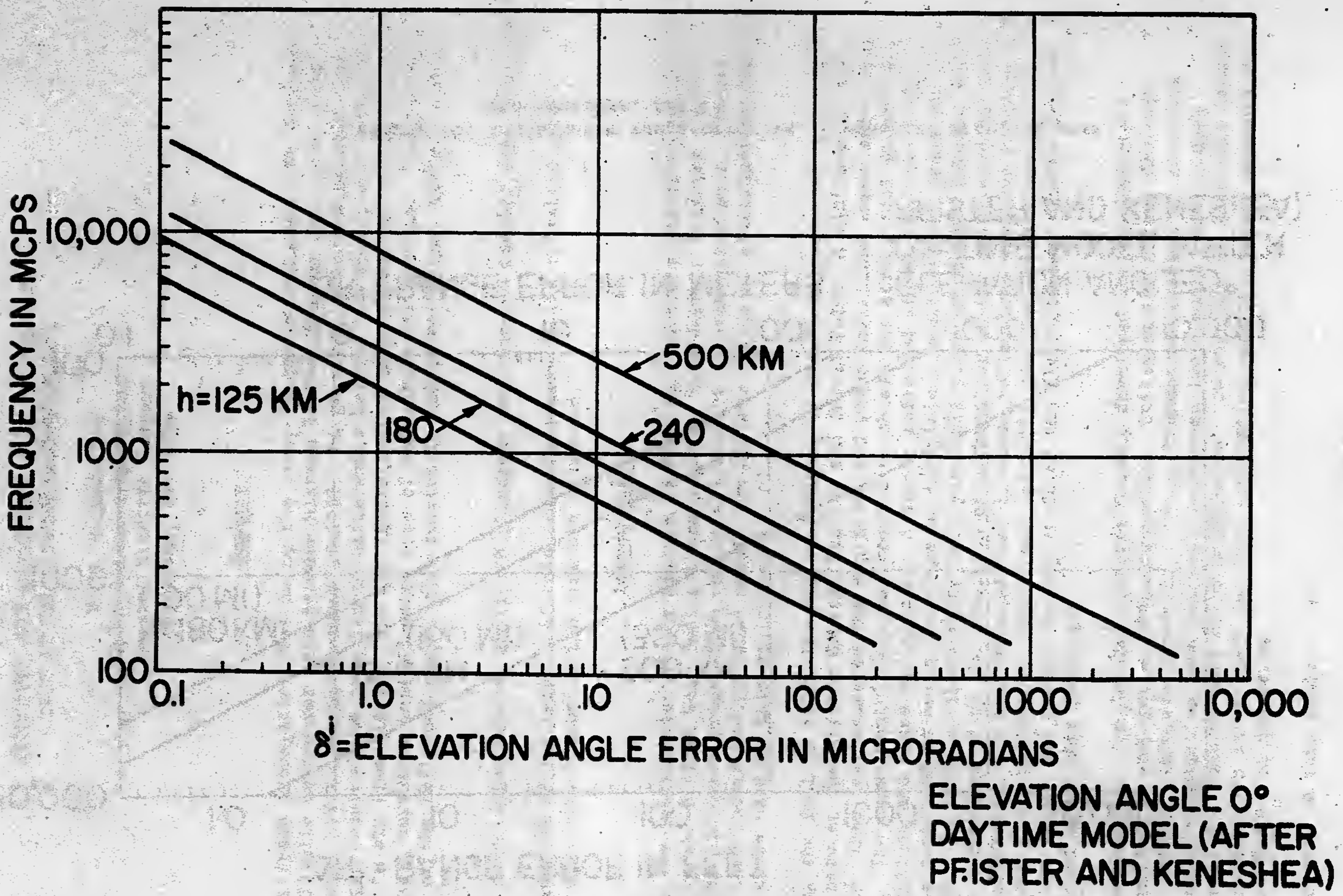


Figure 11. Ionospheric angle error vs. frequency (after Pfister and Keneshea, AFCRL).

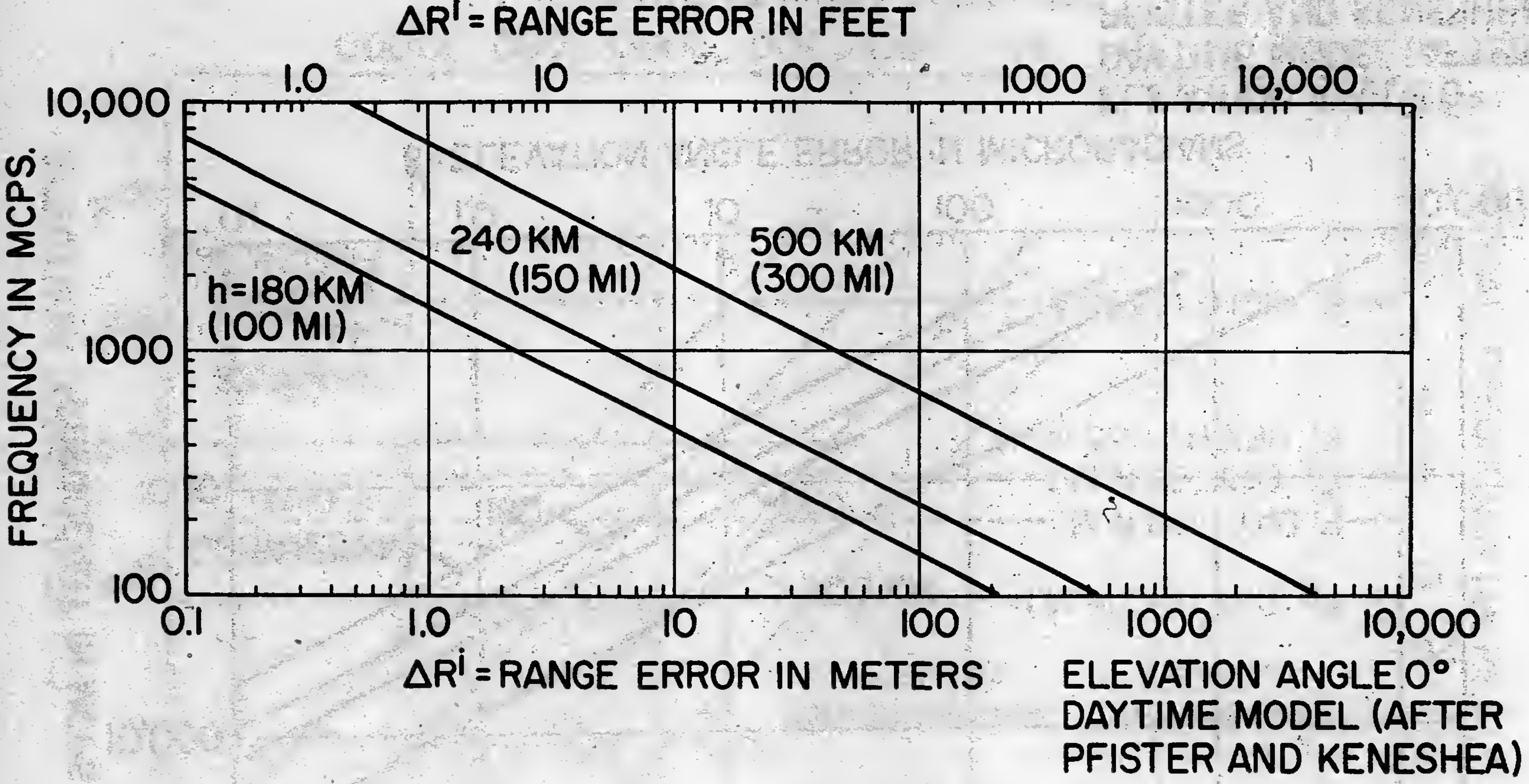


Figure 10. Ionospheric range error vs. frequency (after Pfister and Keneshea, AFCRL).

MICROWAVE OVER-WATER MULTIPATH PROPAGATION TEST AT THE MISTRAM ELEUTHERA SITE

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Summary

A multipath propagation test using the sweep frequency technique was performed over the 28-mile over-water path between Mistram Central Station and Powell Point at Eleuthera, B.W.I. during November of 1963. Large ocean reflected signals were noted. Test results show path length differences between the direct and reflected signal up to six times larger than the theoretically computed values taking into account earth curvature and 4/3 earth type of refraction corrections. The relative magnitude of the reflected signal with respect to the direct signal varied between 0.3 and 0.96.

Poor correlation between multipath phenomena and measured weather data was noted.

Application of the results of this study to improving the reliability of a frequency diversity communication link are discussed.

Introduction

Since the installation of the Mistram II Missile Trajectory Measuring System at Eleuthera in the British West Indies there has been considerable interest in understanding the microwave propagation characteristics of an over-water path. It was felt that this understanding would permit the optimum design of electronic equipment which was to be used in this environment.

During the fall of 1962, the National Bureau of Standards conducted a propagation test between the Mistram Central Station and Powell Point, a distance of about 28 miles.[12] This test measured the amplitude and phase variations experienced over this link. The experimental data was reduced to obtain the statistical properties of the amplitude and phase variations. A strong reflected signal from the ocean surface was suspected as being the cause of the violent amplitude and phase variations noted. In order to investigate this possibility, barge mounted screens were positioned perpendicular to the direction of propagation at the calculated point of specular reflection. It was reasoned that if ocean reflections were the cause of the amplitude and phase variations, then these specially designed screens would block the reflected signal and reduce the amplitude of the fluctuations. Little improvement was noted using this technique.

In the meantime, the General Electric Company developed an instrument for measuring multipath propagation characteristics. Utilizing the latest developments in broadband microwave technology, the instrument was capable of resolving extremely small propagation path differences between two signals; path length differences of 0.1 feet or more could be measured. In the fall of 1963 this equipment was installed to investigate the same propagation path previously measured by the National Bureau of Standards.

The purpose of this paper is to:

1. describe the tests that were performed

2. present the experimental results obtained

3. to demonstrate one use of these results in the optimization of a typical frequency diversity communication link.

Multipath Characteristics

A radio communication link is said to be experiencing multipath transmission if the received signal may be derived from the vector addition of two or more distinct signals, each traveling over its separate path. The general multipath situation that will be discussed, in which the propagating medium produces three distinct paths, is illustrated in Figure 1. The shortest path is arbitrarily labeled the direct signal; it has a relative magnitude of 1.0 and an effective propagation distance, R.

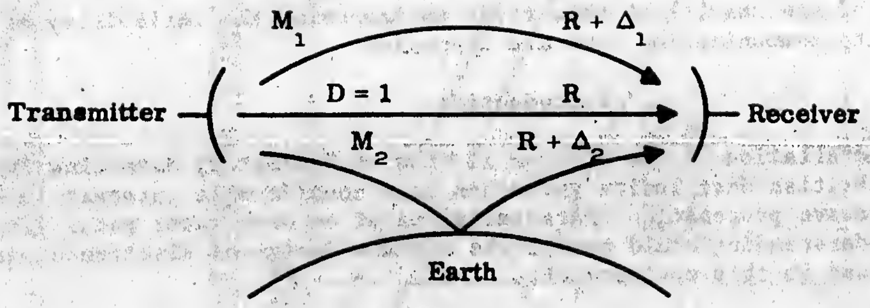


Figure 1. Multipath Transmission with Three Distinct Paths

The upper multipath has a relative magnitude of M_1 and an effective propagation path of $R + \Delta_1$. This multipath is caused by atmospheric anomalies.

The lower multipath has a relative magnitude of M_2 and an effective propagation distance of $R+\Delta_2$. This multipath is caused by reflection from the earth's surface. A 180-degree phase reversal occurs upon reflection at the grazing angles of incidence to be considered. (This is true for both horizontal and vertical polarization.) The effect of the additional path length, Δ, in the propagation path of the multipath signal is to cause an additional phase delay with respect to the direct signal. This phase delay may be expressed as

$$\theta = \frac{2\pi f\Delta}{c} \text{ radians,} \qquad (1)$$

where f is the frequency, Δ is the path length difference, and c is the velocity of propagation. Using the example shown in Figure 1, the received signal can be represented as the vector addition of three vectors as shown in Figure 2.

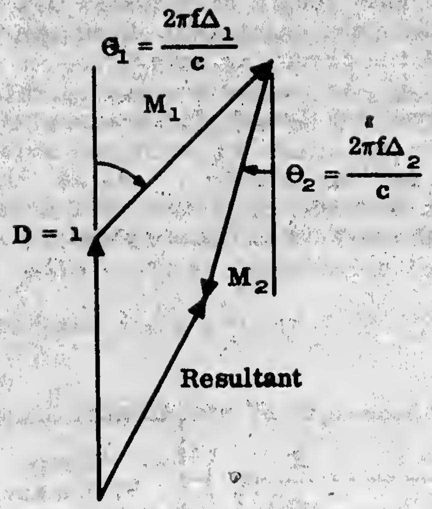

$\theta_1 = \frac{2\pi f\Delta_1}{c}$
M_1
$\theta_2 = \frac{2\pi f\Delta_2}{c}$
D = 1
M_2
Resultant

Figure 2. Vector Addition of Three Vectors

The resultant vector is completely determined from a knowledge of the relative magnitude and additional path lengths of the multipath vectors.

Principle of Measurements

Several techniques may be used to measure the presence of multipath propagation. (See References 1 through 6.) Among the more important techniques are the following:

a. Narrow Pulse Transmission

Multipath propagation would be present if two or more pulses are received for each transmitted pulse.

b. Angle of Arrival Measurements

A narrow-beam antenna is rotated in the vertical plane (similar to the rotation of a height-finding radar). Multipath transmission is present if appreciable energy is received at two or more elevation angles.

c. Sweep Frequency Technique

The transmitter is swept in frequency. In the presence of multipath transmission, the received amplitude versus frequency display will have a characteristic multipath signature that is attributable to the rotation of the multipath vector or vectors, with respect to the direct path, as the frequency changes.

d. Interference Pattern Measurement

A small sampling antenna is used to measure the vertical interference pattern produced by the vector addition of the direct and multipath vectors.

If very small rapid path length differences are to be detected, the sweep frequency technique is the most promising. With the recent development of the backward wave oscillator and the traveling wave tube, it is possible to sweep over a band of about 40 percent of the center frequency, which--if it is centered in the X-band region--would allow path length differences of 0.1 foot and larger to be measured.

The sweep frequency technique is illustrated in Figure 3 for the case of reflections from the ocean surface. For simplicity, only one multipath signal will be considered.

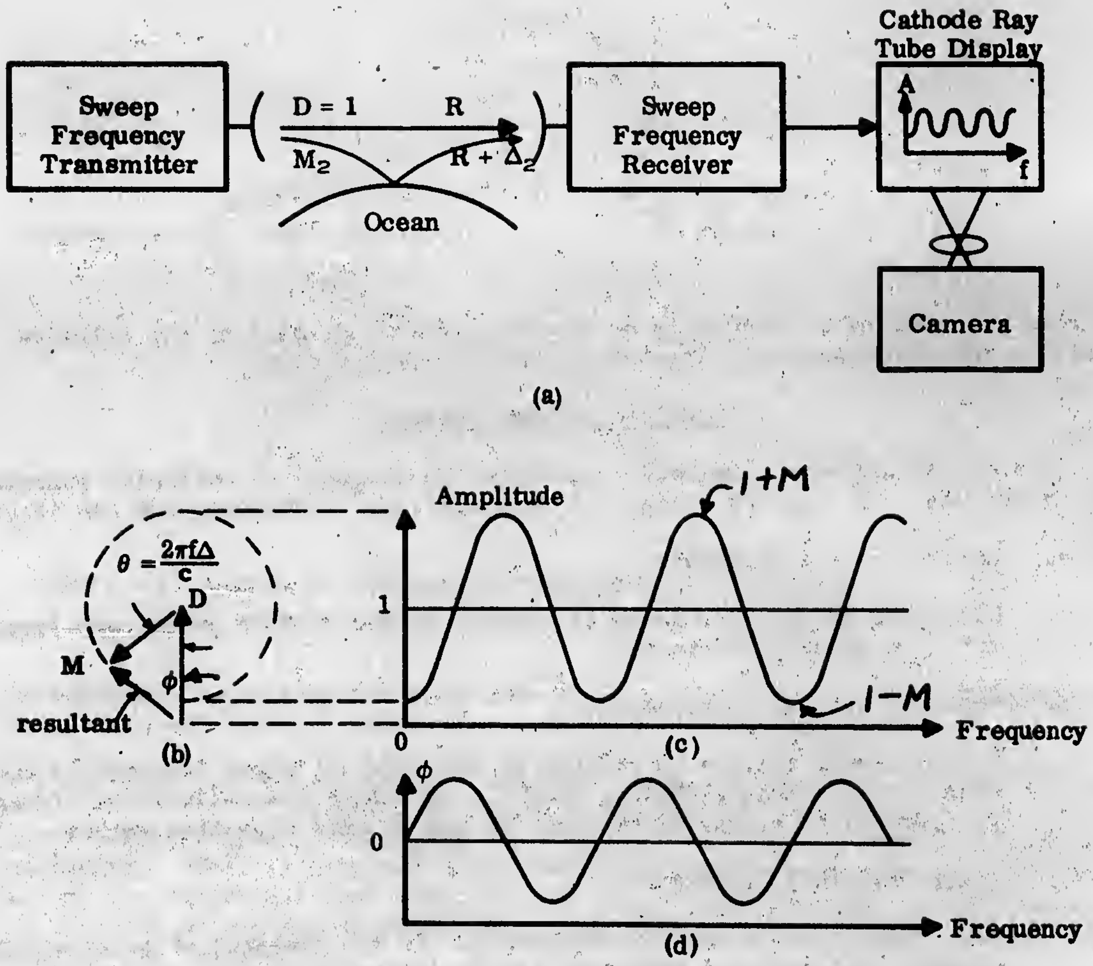


Figure 3 Sweep Frequency Technique

The output of the transmitter is propagated over the multipath environment that is to be measured. The magnitude of the vector resultant of the direct and and multi path signal is received and displayed on the oscilloscope as a function of frequency. From the vector diagram (see Figure 3(b)) it can be seen that as the frequency is increased, the multipath vector will rotate in the clockwise direction through an angle which is directly proportional to frequency.

The corresponding amplitude of the resultant vector is shown in Figure 3(c).

In addition to the amplitude variation, the multipath vector is capable of producing a phase variation as a function of frequency, which is shown in Figure 3(d). The relative magnitude of the multipath signal, M, with respect to the direct signal, D, may be easily calculated from the ratio of the maximum to minimum signal level since the maximum is $1+M$ and the minimum is $1-M$. The additional path length, Δ, may be calculated from the frequency difference between successive nulls. The nulls occur at frequencies for which

$$2n\pi = \frac{2\pi f\Delta}{c} \text{ radians for } n = 0, 1, 2, 3 \qquad (2)$$

so that the path length difference may be expressed as

$$\Delta = \frac{c}{f_\delta} \qquad (3)$$

where f_δ is the frequency between successive nulls.

In the event that two or more multipath signals are present, the complex display of resultant amplitude versus frequency will, in general, require reduction into its individual multipath components by means of a simulator.

Equipment

Transmitter

A simplified block diagram of the transmitter is shown in Figure 4. The transmitter consists of a backward wave oscillator that linearly sweeps between 6.5 kilomegacycles and 11.0 kilomegacycles at a 10-cycle per second rate. The RF output is amplified by a wideband traveling wave tube amplifier and delivered to the antenna through the broadband sampler.

The sampler produces a d-c voltage that is proportional to the RF signal. This voltage is compared to a voltage reference that corresponds to the desired RF output. The difference voltage is amplified and then used to very the backward wave oscillator output in such a way as to maintain the sampler voltage equal to the reference voltage. This feedback arrangement keeps the output signal reasonably constant in spite of the violent power variation that would normally be expected when a backward wave oscillator and traveling wave tube are used over such a wide bandwidth.

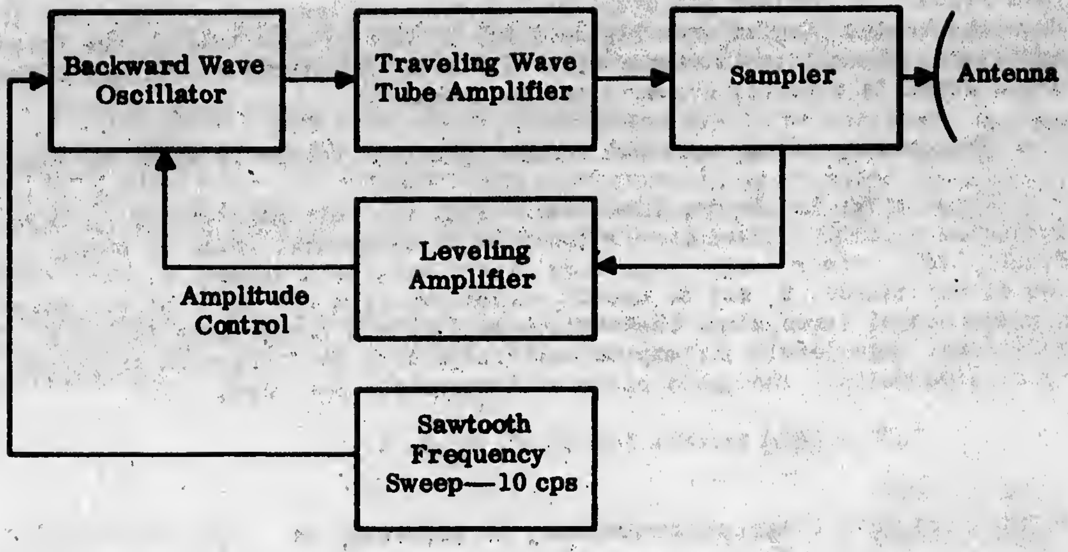


Figure 4   Transmitter, Simplified Block Diagram

Receiver

A simplified block diagram of the receiver is shown in Figure 5. The associated display and recording equipment is shown in simplified block diagram form in Figure 6.

The sweep frequency receiver is designed to follow in frequency the linear frequency sweep of the associated transmitter. It consists essentially of a frequency-lock loop with a sweep assist channel in order to permit the large frequency deviation of the incoming signal to be followed.

The received signal is mixed with the local oscillator in the special broadband mixer located directly behind the tower-mounted receiving antenna. The resultant 30-megacycle intermediate frequency is amplified in a 60-db amplifier, which is also tower-mounted. The amplified signal is transmitted down the tower to the step attenuator, where various fixed attenuations can be switched in the signal path to bring the absolute received level to an acceptable value. Two additional stages of fixed amplification are added before detection. Two detectors are available. One is a linear detector which produces a low-frequency signal output that is linearly proportional to the intermediate frequency signal. The other is a logarithmic detector which has the capability of expanding the lower signal levels. All the data analyzed in this report was obtained with the linear detector.

A frequency-lock loop with sweep assist is used to obtain accurate tracking of the local oscillator.

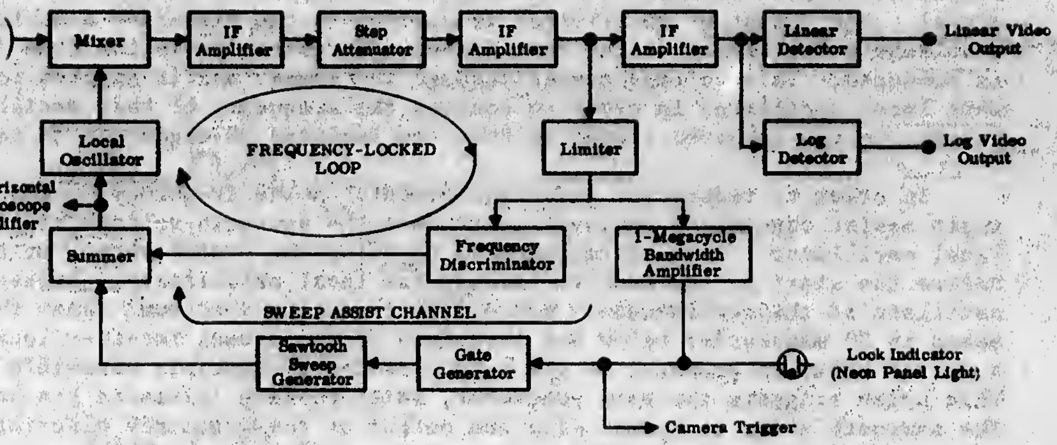


Figure 5 Sweep Frequency Receiver, Simplified Block Diagram

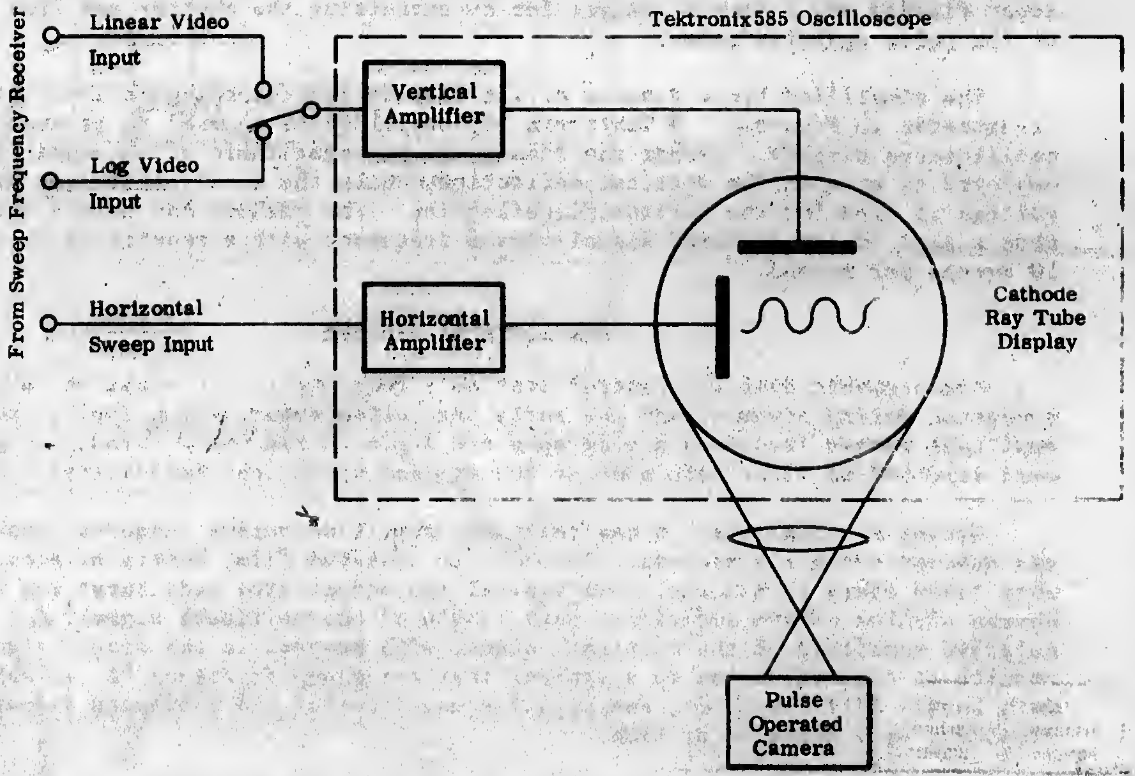


Figure 6 Display and Recording Equipment, Simplified Block Diagram

A portion of the IF signal is hard-limited in order to obtain a signal output for which the amplitude is constant irrespective of the level of the input signal. One output of the limiter is applied to the frequency discriminator where a voltage proportional to the instantaneous frequency deviation from 30 megacycles is produced. This voltage passes through the summer and is applied to the backward wave local oscillator in order to control the frequency of this oscillator with respect to the received signal so that the required 30-megacycle IF is produced.

In order to reduce the gain requirements of the frequency-locked loop, a sweep assist channel is employed to produce the sawtooth voltage required by the local oscillator. This is implemented by using the second output of the limiter. Before the start of the receiver sweep, the local oscillator is preset to oscillate at the low-frequency end of the frequency spectrum. When the transmitter sweep is 30 megacycles below the frequency of the preset receiver local oscillator, a narrow pulse is produced at the output of the 1-megacycle bandwidth amplifier. This pulse triggers the gate generator, which in turn initiates the sawtooth. The sawtooth sweep is summed with the output of the frequency discriminator and applied to the local oscillator in order to produce the predominant portion of the frequency sweep voltage. The frequency-locked loop signal then corresponds to the error between the transmitter and receiver sawtooths.

In addition to the two video output signals, the receiver provides a sawtooth signal that is proportional to frequency for horizontal deflection of the oscilloscope display and a camera trigger for synchronizing the shutter and film pull-down of the pulse-operated camera.

The simplified block diagram of the display and recording equipment is illustrated in Figure 6. A Tektronix 585 oscilloscope is used to produce the oscilloscope display. Either the linear or the logarithmic video signal is employed to produce the vertical deflection, while the local oscillator sweep voltage is used for the horizontal deflection. The cathode ray tube display is then a plot of the received signal-versus-frequency with a repetition rate of 10 sweeps per second.

Experimental Results

Measurements over the Central Station - Powell Point propagation path were conducted during November and the early part of December 1963. Typical resultant amplitude versus frequency photographs are shown in Figures 7 through 10 which were selected to illustrate some of the extreme conditions encountered.

During a normal test, a new resultant amplitude versus frequency photograph was exposed every 1.6 seconds. In order to conserve film, bursts of about 10 frames were taken every 15 minutes. One typical photograph from each burst was reduced to obtain a value of the additional path length of the multipath signal, Δ , and the relative magnitude of the multipath signal with respect to the direct signal. The results of this reduction on a typical test are shown in Figure 11, in which the path length difference, Δ , and relative magnitude of the multipath signal are displayed as a function of time.

The dominant multipath signal was produced by ocean reflection. The path length difference varies between 0.09 and 0.4 foot, and the relative magnitude varied between 0.3 and 0.96. The larger values of relative magnitude are difficult to read accurately and it is therefore probable that values of relative magnitude larger than 0.96 were experienced from time to time. Theoretically, the reflected

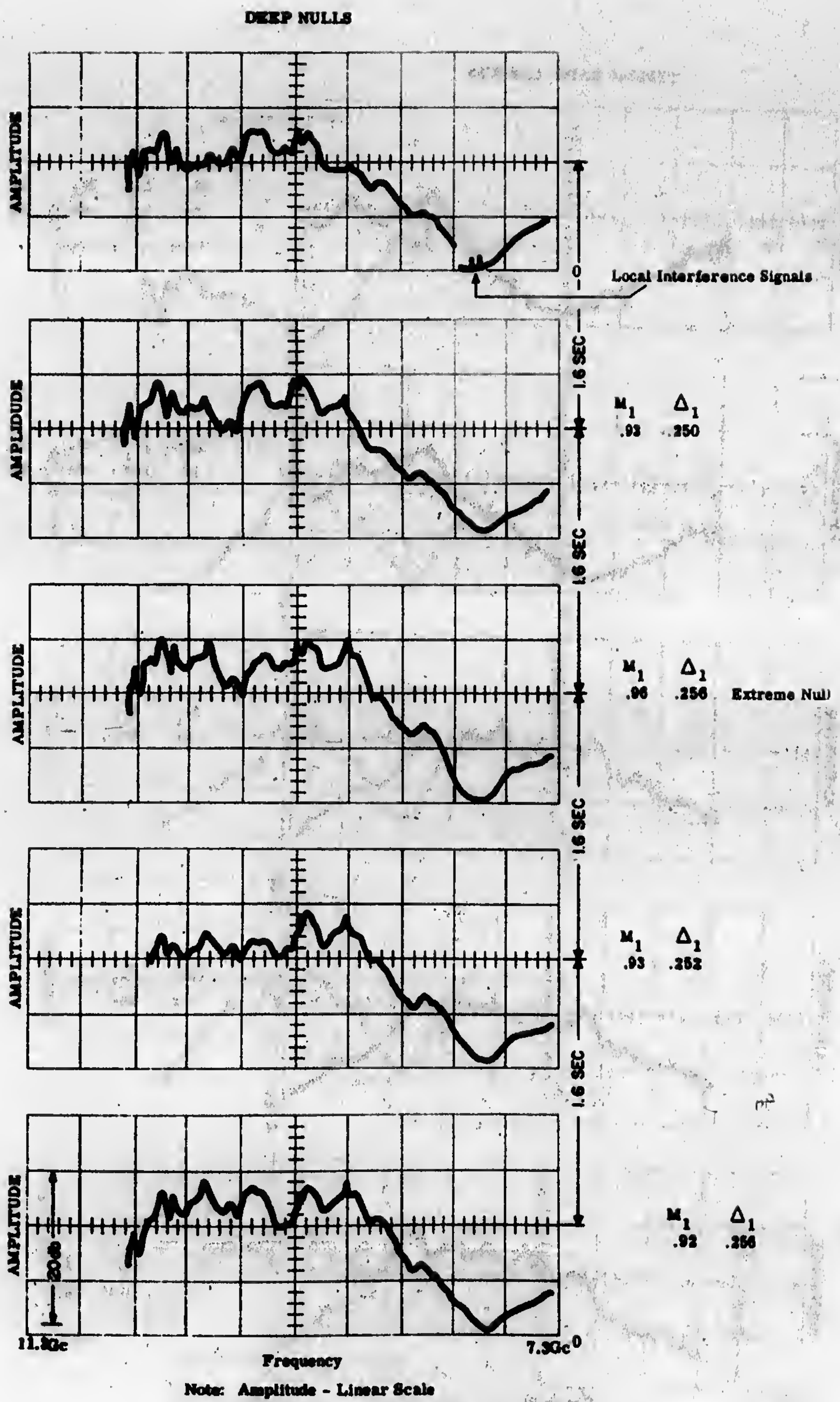
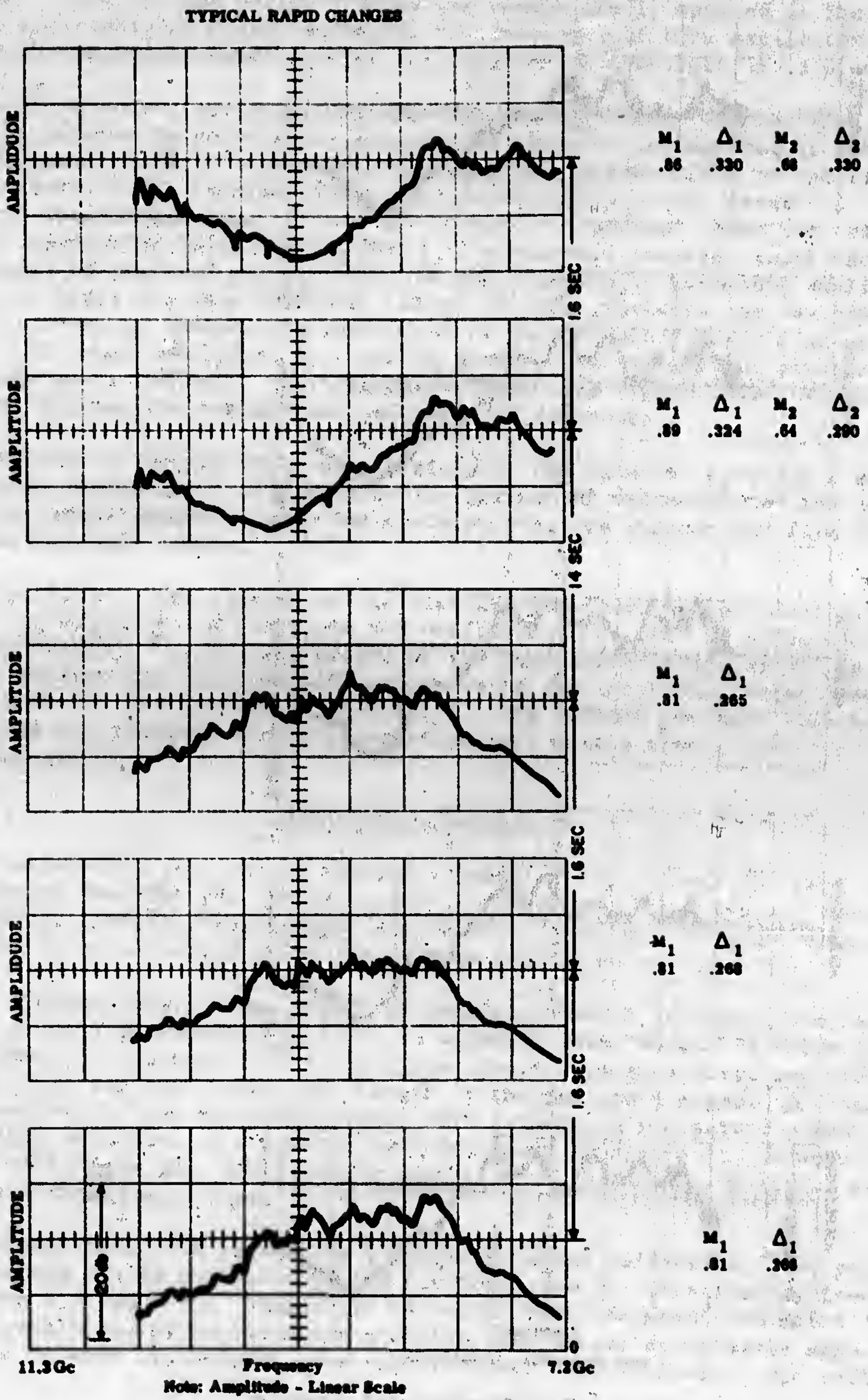


Figure 7 - Experimental Results



Note: Amplitude - Linear Scale

Figure 8 - Experimental Results

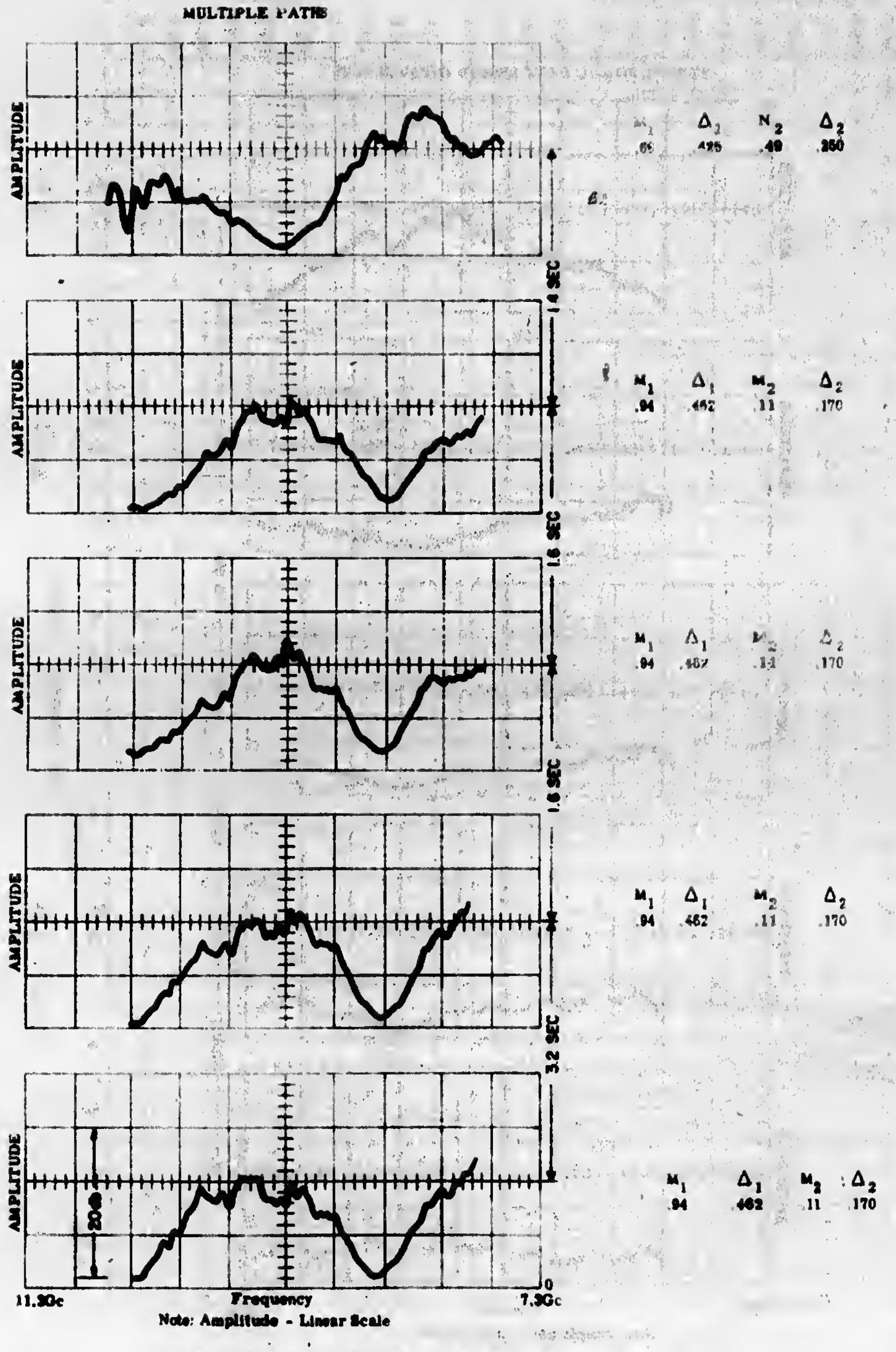
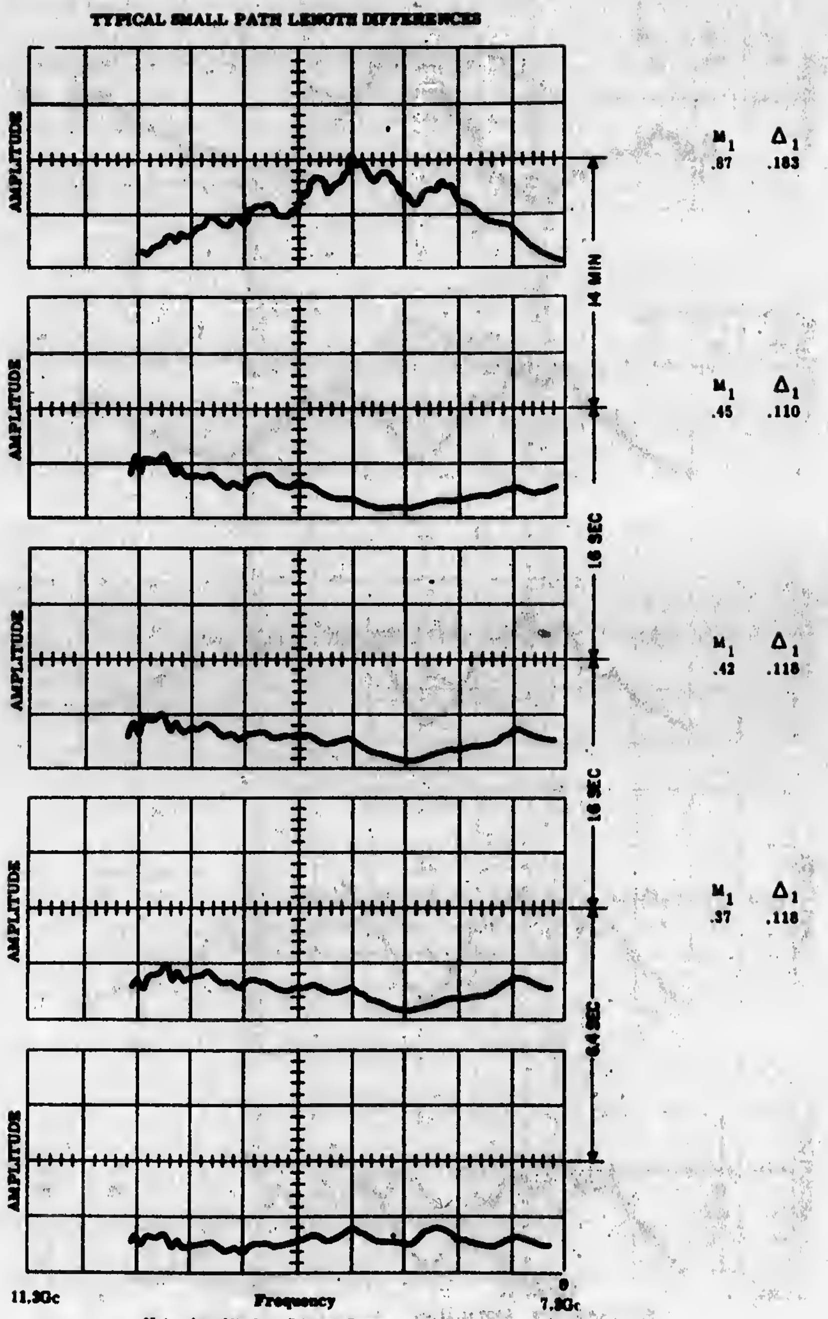


Figure 9 - Experimental Results



Note: Amplitude - Linear Scale

Figure 10 - Experimental Results

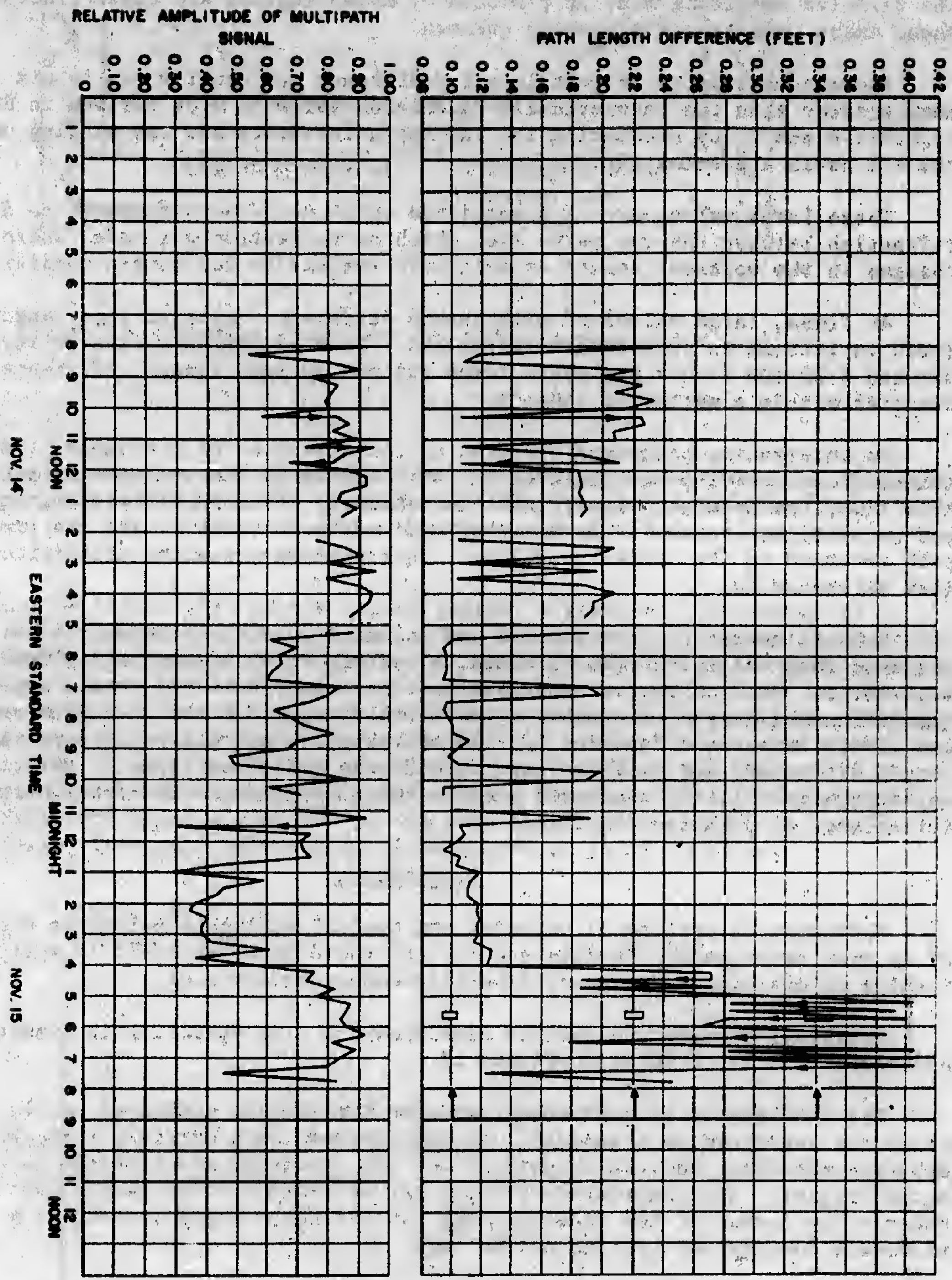


Figure 11

- Typical Path Length Difference, Δ , and Relative Amplitude versus Time signal should be reduced from unity by the value of the divergence factor which is calculated in Reference 15, Appendix A to be about 0.52. This average magnitude is in approximate agreement with theoretical calculations. The wide variations in the relative magnitude must be produced by meteorlogical and ocean fluctuations whose exact origins are presently unknown.

The measured value of path length difference was usually (up to six times) much greater than the theoretically calculated value of 0.06 derived in Reference 15, Appendix B and C, by accounting for the earth curvature and ray bending using the 4/3 earth approximation.

These large values are of a magnitude which could be attributed to differential refraction between the two paths (for which no correction was made) and/or due to changes in the apparent center of multipath reflection due to sea conditions.

At times, large values of path length difference occurred simultaneously with rapid variations in path length difference. This is well illustrated in Figure 11 between 5:00 and 7:00 A.M., where large changes in path length differences often occurred within a matter of seconds.

An interesting phenomena occurred on the afternoon of 6 November 1963 when thunderstorm activity was observed in the vicinity of the propagation path. At this time, the absolute signal level increased by about 20 db at all frequencies and no multipath transmission was observed. After several hours, the propagation path returned to its normal condition. This phenomenon is characteristic of ducting (see Reference 10).

A large amount of weather data was accumulated during the multipath tests in the hope that multipath results could be correlated with some meteorlogical characteristic. Part of the recorded weather data was reduced to obtain some derived quantities which were also used in the correlation. All the available weather data was then plotted as a function of time and visually correlated with plots of path length difference and relative magnitude of the reflected signal. Unfortunately, not enough correlation was noted between the quantities to be of any help in predicting poor propagation conditions.

Application

The previous section illustrated the typical multipath parameter variations of an over water path. In this section, it would be of interest to apply these results to optimize a frequency diversity communication link.

A typical frequency diversity communication link operating in a multipath environment is illustrated in Figure 12.

The information to be transmitted over the link is modulated on two carriers which are separated in frequency. At the terminal each carrier is received by a separate receiver, and the outputs of the two receivers are combined to yield one output signal. The combination process may be produced by a simple switch which selects the output of the receiver with the strongest input signal, or a more elaborate combination process may be used.

For illustrative purposes in this paper let us consider that a simple switch is used to select the receiver with larger input signal.

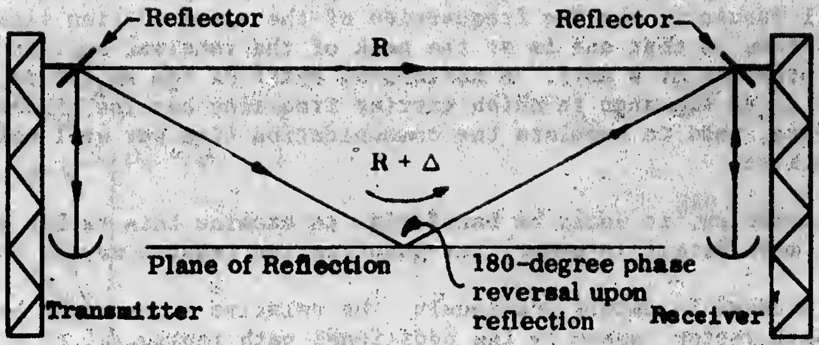


Figure 12 - Typical Frequency Diversity Communication Link

In a frequency diversity communication link, it is reasoned that the propagation conditions experienced by the two separate carriers frequencies are sufficiently different so that the probability of both carriers simultaneously experiencing deep fading conditions is extremely small.

For carrier frequencies in the X-band region, a frequency separation of between 2 and 4 percent would be typical. This corresponds to frequency separation of about 200 MC. In general, this degree of frequency separation is usually sufficient to produce uncorrelated noise at the receiver, when this noise is generated in the atmosphere. It would now be of interest to calculate the performance of our typical frequency diversity communication link in an ocean reflected multipath environment. As illustrated by the experimented data in Figure 7 through 11, the presence of the reflected signal causes severe fading at certain frequencies, the exact frequency being a function of the path length difference. A typical amplitude versus frequency spectrum is illustrated in Figure 13 below.

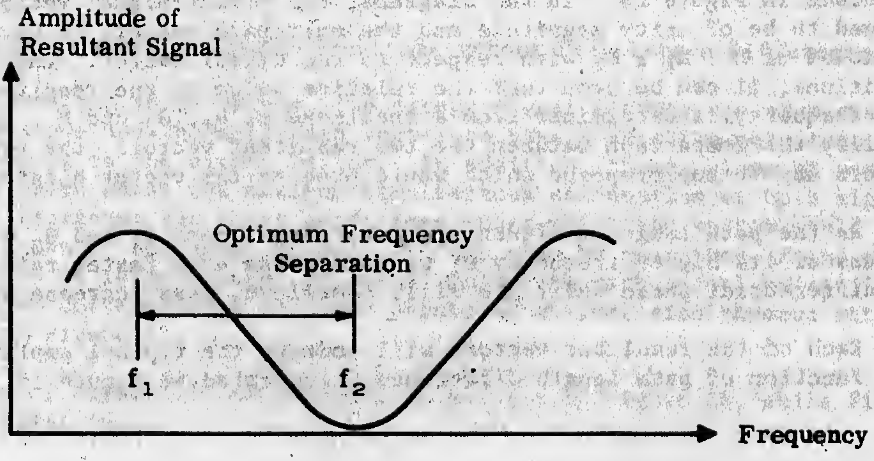


Figure 13 - Typical Amplitude versus Frequency Display

From this figure, it can be concluded that the optimum frequency diversity action will result if the two frequencies of the communication link are positioned in the spectrum so that one is at the peak of the received amplitude spectrum while the other is at a null. A horizontal shift of the amplitude spectrum curve may result in an exchange in which carrier frequency has the higher amplitude and is, therefore, used to complete the communication link but will not cause communication failure.

In proceeding, it would be beneficial to examine this rather obvious conclusion in more detail in order that quantitative results may be obtained.

As discussed in detail previously, the relative phase angle between the direct and multipath vector caused by the additional path length, Δ, of the reflected signal is given as:

$$\theta = \frac{360 f \Delta}{C} \text{ degrees.}$$

where f is the operating frequency and C is the velocity of light. Because of the slight difference in carrier frequencies, the phase shift produced by multipath at each carrier frequency will be slightly different. In particular, let the phase shift produced at the higher frequency f_1 be θ_1 and that at the lower frequency, f_2 by θ_2. In addition, let the path length difference, Δ ; be common for both frequencies. The vector diagram corresponding to the two received signals

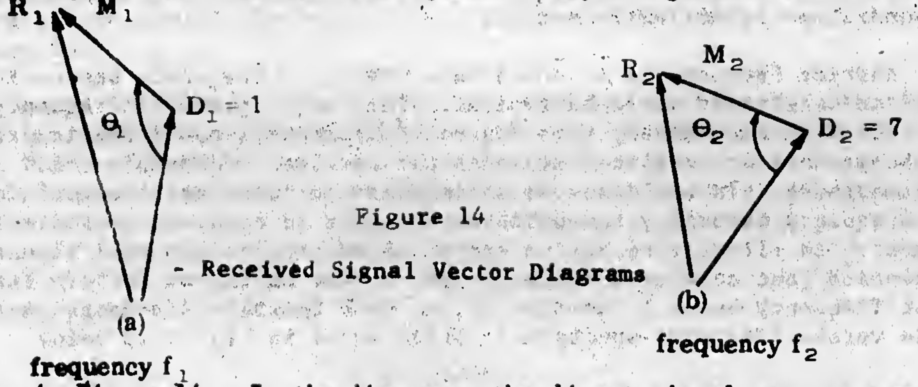
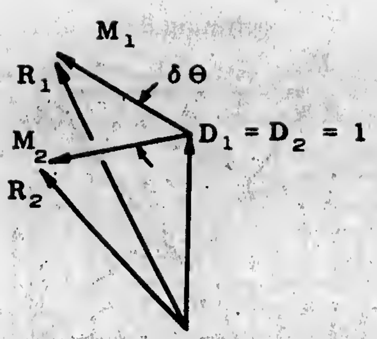


Figure 14 - Received Signal Vector Diagrams

are drawn in Figure 14. In the diagrams, the direct-signal vectors are each assumed to be of unity magnitude and the multipath vectors to have a relative magnitude of $M = M_1 = M_2$ with respect to the direct vector. Under the above conditions, it can be seen that the relative value of the resultant vector at each frequency is determined from a knowledge of θ_1, θ_2 and M. In order to simplify the comparison between the two resultant vectors, the two direct signal vectors may be superimposed which would produce the vector diagram of Figure 15.

As the path length difference, Δ, increases, both θ_1 and θ_2 will increase. Because of its higher frequency θ_1 will increase at a faster rate than θ_2 and the differential phase shift $\delta\theta$ will, therefore, also increase as a function of Δ

Each of the resultant vectors will undergo the typical amplitude pertubations as a function of path length difference illustrated in Figure 16.



$$\Theta_1 = \frac{360 f_1 \Delta}{C}$$

$$\Theta_2 = \frac{360 f_2 \Delta}{C}$$

$$\delta\Theta = \frac{360\,\delta f\,\Delta}{C}$$

where

$$\delta f = f_1 - f_2$$

Figure 15 - Superposition of Received Signal Vectors

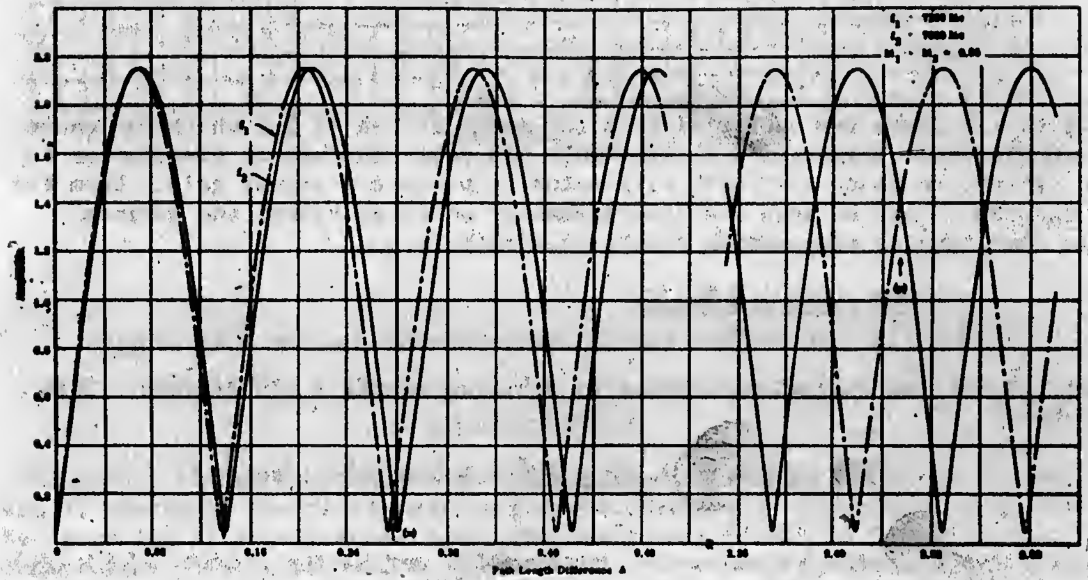


Figure 16 - Received Signal Amplitude versus Path Length Difference

In generating these curves, the direct signal vectors are each assumed to be of unity magnitude and the multipath vector to have a relative magnitude of M = .95 with respect to the direct vector. In addition, those values of path length difference, Δ , which cause the maximum simultaneous attenuation of both signals and which would represent the worst conditions under which the system would be required to operate are shown by the small arrows. It can also be seen that if the path lengths difference is large enough, an actual multipath path signal gain will occur. The vector diagram corresponding to those points of maximum simultaneous attenuation is illustrated in Figure 17.

The diagram of Figure 17a corresponds to point x of Figure 16, while Figure 17b corresponds to point (y).

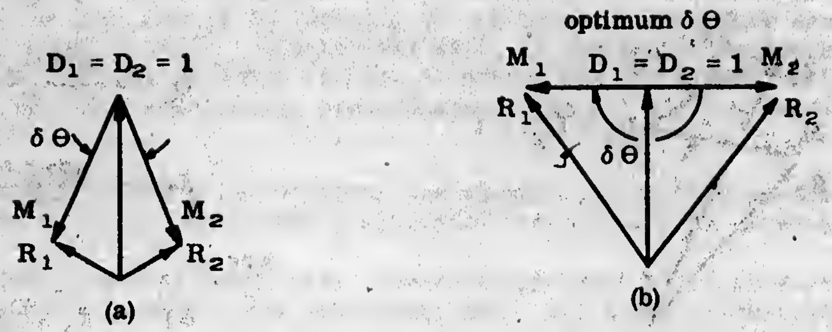


Figure 17 - Vector Diagrams for Poor (a) and Optimum (b) Frequency Diversity

Figure 17a shows the vector diagram for small values of $\delta\theta$ which can cause severe simultaneous attenuation while Figure 17b shows the vector diagram for an optimum system, in which multipath will actually produce a signal gain. From the vector diagram it can be seen that the parameter which describes the maximum possible simultaneous attenuation due to multipath is:

$$\delta\theta = \frac{360 \quad (\delta f)(\Delta)}{c}$$

and that the optimum system performance is achieved when $\delta\theta = 180$ degrees. This occurs when:

$$(\delta f)\ (\Delta) = \frac{c}{2} = \frac{9.8 \times 10^8}{2} = 4.9 \times 10^8 \text{ ft/sec}$$

A typical curve of maximum simultaneous attenuation versus the product $(\delta f)\ (\Delta)$ is shown in Figure 18. A value of $M = .95$ was used in constructing this graph.

The peak in this curve illustrates the optimum value of the product of the differential frequency and the path length difference. In practice, the value of Δ may be expected to vary due to various meteorlogical conditions. In order to obtain a truly optimum system in a practical situation, the span of values of Δ actually experienced should be used in calculating the optimum frequency separation between channels.

It can be seen that the peak to the curve is rather broad so that the reasonable variations in Δ due to meteorlogical changes should produce only moderate degradation of system performance.

An optimum system, in the multipath sense, would have a differential frequency wavelength, $\lambda_{\delta f} = \frac{c}{\delta f}$, which is twice the path length difference. The path length difference would be calculated by taking the average of the maximum and minimum path length difference encountered.

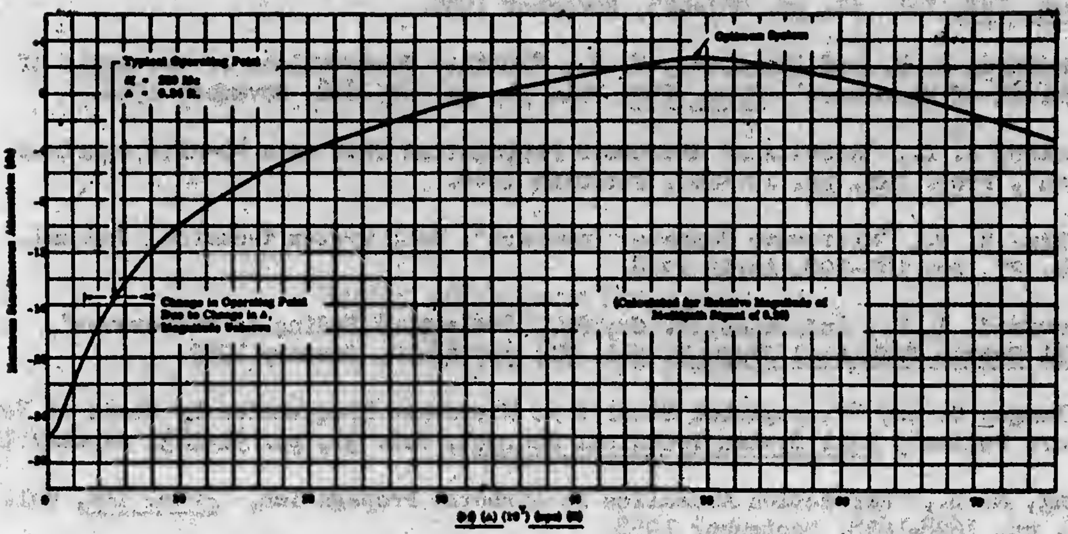


Figure 18 - Maximum Simultaneous Attenuation versus δf) (Δ)

Acknowledgement

The author wishes to acknowledge the outstanding efforts of Mr. J. J. Gaura who was in charge of the field testing phase at Eleuthera and Antenna Park, and of the reduction of experimental data; the assistance of Mr. G. Tucker in the testing phase at Eleuthera, Mr. A. LaMacchia in the reduction of the experimental data.

Special mention should be made of the support given to this project by Lt. John Cribbs, USAF, AMR, who clearly saw the need for a test of this nature and supported the effort through the many obstacles encountered.

The excellent cooperation given in the field testing portion of this program by both Pan American and RCA personnel is sincerely appreciated.

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January, 1964

Random Atmospheric Propagation Noise in Range Rate Difference Data While Tracking a Moving Target

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Summary

Recent low elevation angle space missions have provided a means of detecting random atmospheric propagation noise in range rate difference radar data while tracking a moving target.

The range rate difference data noise increases as the elevation angle approaches the horizon. The noise increases as an approximate function of $\dot{E} \csc^2 E$ down to about 4 degrees elevation and increases less rapidly below 4 degrees.

Introduction

Data noise is very significant while radar tracking a missile or satellite when it is necessary to make real time corrections. The data processing equations must be designed to make timely and accurate computations in spite of the noise on the data. Therefore, the characteristics of this noise must be understood so that the proper allowances can be made.

The General Electric Mod III radio guidance system tracked and guided the launch vehicles of the highly successful Mercury Program. During these and other low elevation angle missions, propagation effects were noticed in the range rate difference* data. An intensive investigation of these propagation effects is continuing. Some of the pertinent results of this investigation are included in this report.

This report presents some of the "raw" radar data noise, suggests a curve fitting method based upon mechanism considerations, and attempts to determine the significance and implications of this curve fitting.

Random Atmospheric Noise in Range Rate Differences

Any theory as to the method of reducing the effects or determining the cause of atmospheric propagation noise must take into consideration the observed characteristics of this noise. Much data is available on propagation effects while tracking a stationary target. However, data showing propagation effects while tracking a moving target is not widely published. Therefore, Figure 1 presents samples of propagation noise as observed in radar data obtained during four representative low elevation angle missile flights.

A better understanding of the information in this report is obtained by knowing and understanding the terms and concepts used. These are as follows: The noise on the radar signals is observed as noise in the radar data. Therefore, the noise analyzed in this report is radar data noise. The magnitude of the data noise is obtained by smoothing the raw radar data and then computing the standard deviation of the smoothing residuals, using a fixed number of residuals for each computation. Careful selection of filters ensures efficient removal of noise without erroneously smoothing out missile steering. The adequacy of these processes has been demonstrated by comparing powered flight and free flight noise estimates. Although more sophisticated noise determination methods may be developed, there is no reason to believe that these methods will produce results which significantly differ from those already obtained.

In Figure 1, the radar data noise has been plotted as a function of elevation angel. (Elevation angle is measured up from the horizon.) The decision to plot noise as a function of elevation angle results from the consistency with which this method can be used to compare the noise of different missile flights. Often weapon system missile flights exceed the radar range, altitude and flight time of the low elevation angle missions, without random propagation noise being observed in any significant amount. The noise has been normalized by dividing by a constant for the following reasons:

1) differing radar sampling rates, sampling duration and smoothing techniques change the apparent noise level,
2) the relative changes of noise are more significant than the absolute changes, and
3) to avoid security problems.

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* Range rate differences (also called lateral rates) are defined as the difference between the line-of-sight range rate observed at one antenna and the line-of-sight range rate observed at another antenna. The second antenna is located at some base line distance from the first antenna.

The crux of understanding propagation noise while tracking a moving target is appreciation of the dependence of noise magnitude upon elevation angle. Determining this relationship and interpreting the significance of this relationship is the purpose of the remainder of this paper.

Curve Fitting Propagation Noise Based on Mechanism Considerations

To simplify the presentation of noise data, facilitate generalizations, permit extrapolations, and most significant, provide an insight into the noise mechanism, it is necessary to determine some analytical relationship between noise and elevation angle. Whereas the atmosphere is the cause of propagation noise, it is natural to determine various relationships of the radar beam in the atmosphere.

Figure 2 is representative of the efforts to determine which relationship between the radar beam and the atmosphere better describes the observed noise. (To facilitate comparisons, the data has been normalized to unity at 10°.) The small ovals of Figure 2 represent the actual noise values from a typical low elevation angle mission. Two attempts were made to fit this data.

The first attempt resulted from assuming that the noise is proportional to the length of the radar beam in the atmosphere. This radar beamlength would be calculated to some effective atmospheric altitude (A). If a flat earth is assumed, this length of the radar beam is $A \csc E$. The normalized (to unity at 10°) $A \csc E$ curve is illustrated in Figure 2. The curve does not fit the actual noise values depicted in Figure 2, therefore, it seems that the length of the beam in the atmosphere is not the most significant propagation noise factor.

The second attempt results from assuming that the noise is proportional to the horizontal velocity of a moving radar beam along some effective atmospheric altitude A. If a flat earth is assumed, this horizontal velocity of the radar beam is $A \dot{E} \csc^2 E$ (see Figure 3 for the derivation). A normalized (to unit at 10°) $A \dot{E} \csc^2 E$ curve is illustrated in Figure 2, and is depicted in Figure 2 by a solid line. The curve does fit the actual noise values depicted in Figure 2, therefore, it seems that the horizontal velocity of the beam along some effective altitude is a significant propagation noise factor.

Figure 2 is a somewhat simplified presentation. A larger sample of data is available from each missile flight, and about a dozen applicable flights are available. Therefore, to better verify this data fitting technique another but more complete illustration has been prepared. This new illustration presents the data in a form which enables a simpler check on the validity of $\dot{E} \csc^2 E$ as a noise model. If $\dot{E} \csc^2 E$ represents the rate of noise change, then division of the raw noise by $\dot{E} \csc^2 E$ should produce a scatter of points which are independent of elevation angle and is therefore, a noise model validity check. (For reasons previously mentioned, the raw noise is normalized by dividing by a constant "K.")

Figure 4 presents the data noise statistics from two flights after division by $K \dot{E} \csc^2 E$. The resultant scatter of points is fairly independent of elevation angle. Therefore, this indicates that the noise model is valid.

However, as Figures 5 and 6 indicate, not all low elevation angle missile flights can be fitted this well. The top half of Figure 5 presents the quotient of noise divided by $E \csc^2 E$. This quotient increases as E decreases (until about 4 degrees). This indicates that a power of csc E higher than 2 might be used to produce a scatter of points fairly independent of elevation angles. Figure 6 presents a case where division by $\dot{E} \csc^2 E$ seems to overcompensate the dependence on elevation angle and, therefore, the resultant decreases as E decreases. This suggests a power of csc E less than 2 might be used to produce a scatter of points fairly independent of E. Fitting the data to a variable powered csc E term would produce a scatter of points independent of E in most of the cases. However, it is more difficult to imagine a noise mechanism based upon a variable powered csc E. Also, the discrepancy is not as bad as one might first assume. The noise model of C has a spread of a factor of about 3 from minimum to maximum whereas the noise data has a much larger spread of a factor of about

25. Likewise, the noise model of D has a spread of a factor of 2, but the data noise has a greater spread of a factor of 4 from minimum to maximum. Thus, a considerable reduction in spread has been achieved by using $\dot{E} \csc^2 E$, and a closer fit by using a variable power csc E is not a necessity.

Interpretation of the Curve Fit

Generalizations and extrapolations can be made from the curves which are fit to the observed noise. However, these generalizations and extrapolations must be qualified. If $A \dot{E} \csc^2 E$ does represent the horizontal beam velocity which, in turn, represents the noise mechanism, then it is necessary to determine the conditions under which $A \dot{E} \csc^2 E$ is valid.

If $\dot{E}$ is zero, then the equation implies that no propagation noise is to be expected. However, extensive work by the National Bureau of Standards in tracking stationary targets indicates that this is not true. This apparent discrepancy is probably the result of the "stationary target" noise being insignificant in comparison with equipment noise and "moving target" noise.

The equation $A \dot{E} \csc^2 E$ is derived for a flat earth and goes to infinity when E goes to 0 degree. This equation is, therefore, inaccurate at low elevation angles and should be replaced by an equation derived for a spherical earth. The horizontal (arc) velocity of a radar beam along layers around a spherical earth is given by:

$$\text{horz. vel.} = \dot{E}\,(R + A)\left[1 - \frac{R \sin E}{(R + A)\sqrt{1 - \left(\frac{R \cos E}{R + A}\right)^2}}\right]$$

where:

R = earth radius
A = altitude of the effective atmosphere
E = elevation angle of target, up from horizon
$\dot{E}$ = elevation angle rate

Figure 7 illustrates the divergence of flat earth effects (the $\csc^2 E$ curve) from spherical earth effects (the curves which are a function of A and E). This effect of noise increasing less rapidly than $\csc^2 E$ (as E decreases) has been observed. From Figure 5, comparisons can be made of the effects of assuming a flat earth with the effects of assuming a spherical earth. Below 4 degrees for a flat earth (in Figure 5), the $\csc^2 E$ increases much more rapidly than the magnitude of the data noise. By assuming a spherical earth and an effective atmospheric altitude of 21,500 feet (suggested by analysis of radiosonde information), the divergence of the values at the low elevation angles is significantly reduced.

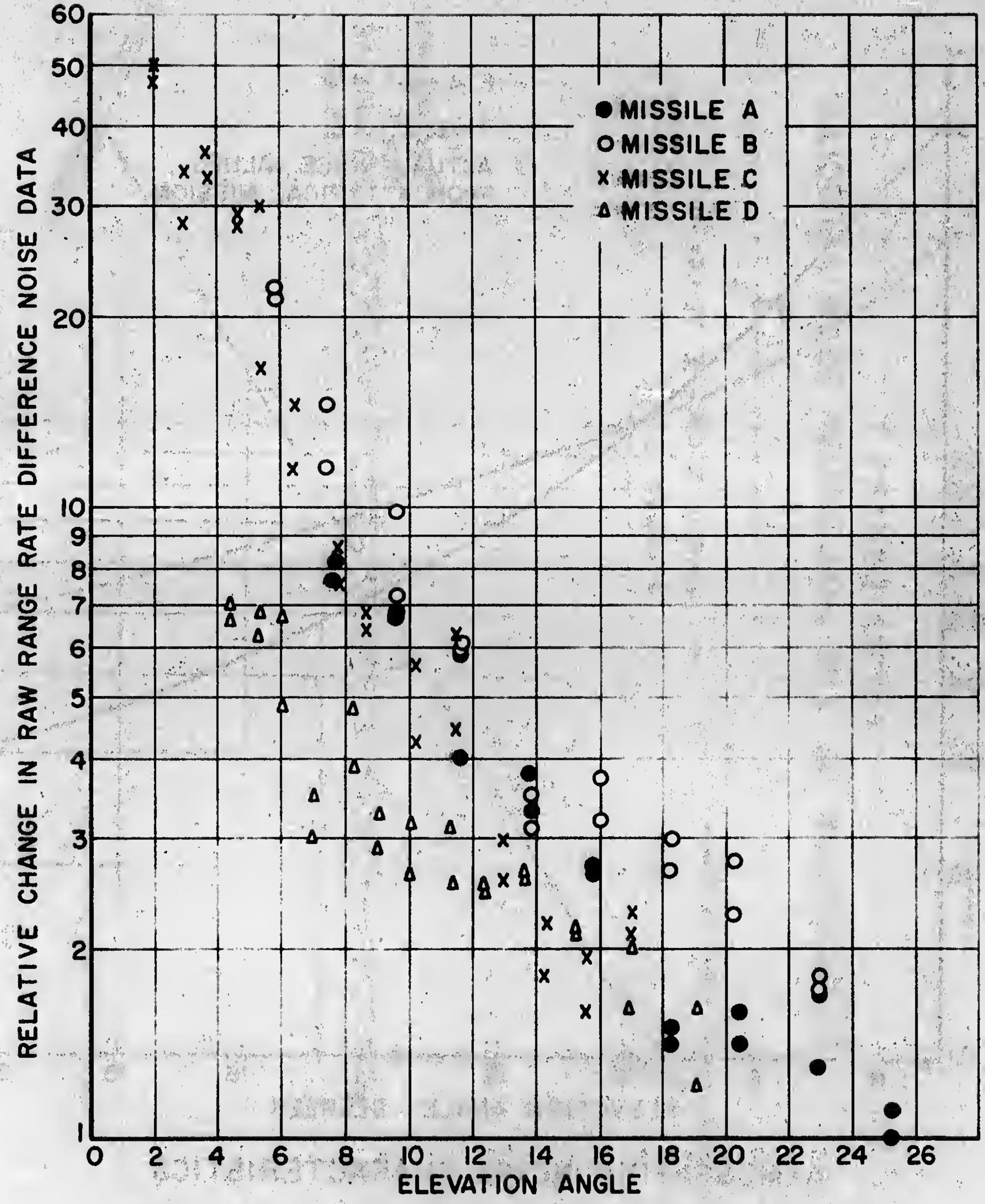
Conclusions

Atmospheric phenomena cause noise in range rate difference radars which increases as a function of $\dot{E} \csc^2 E$ down to 4 degrees elevation angle, and more slowly below 4 degrees.

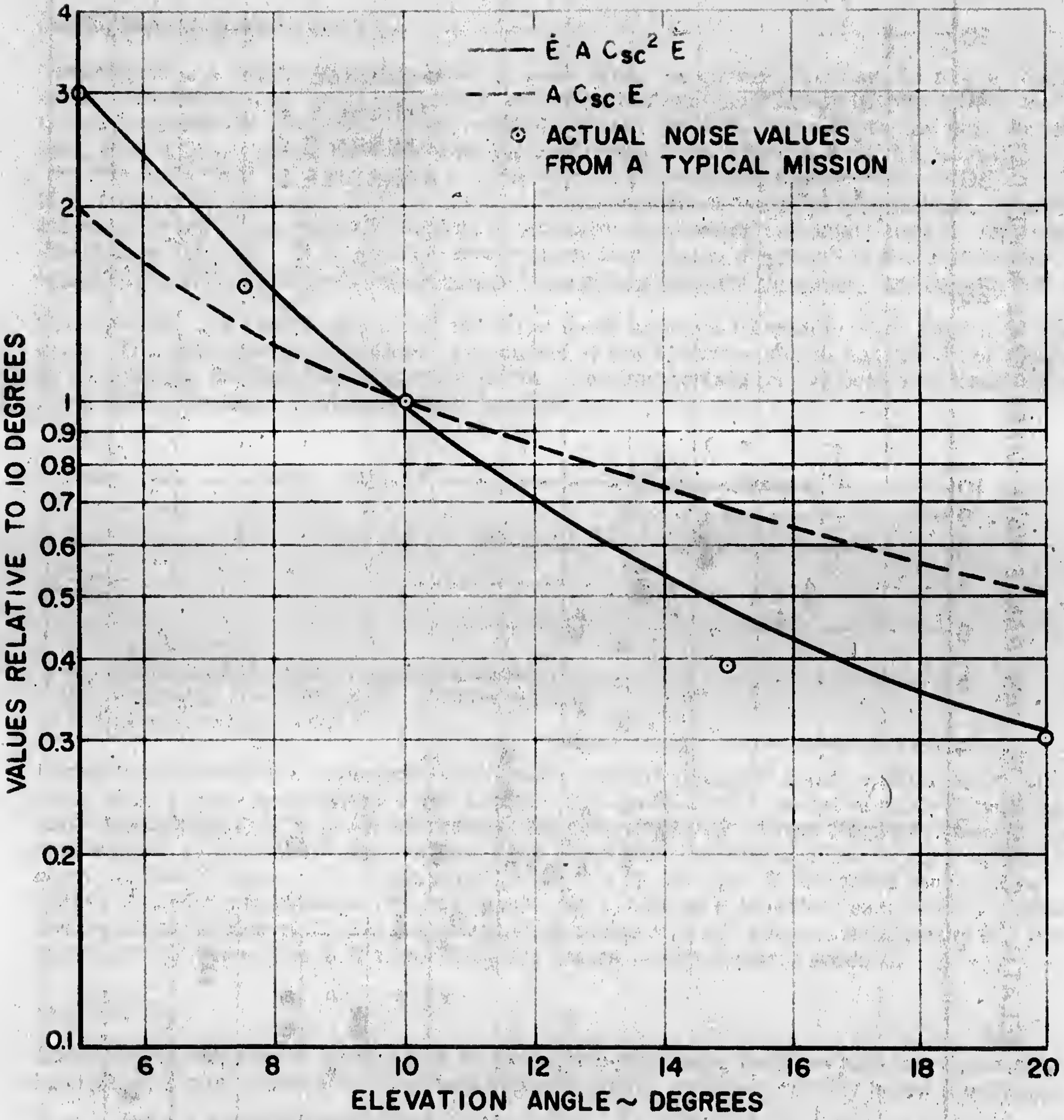
The magnitude of phase noise in range rate difference radars can be estimated by knowing the missile trajectory.

Once atmospheric noise is detected and measured by a radar, its magnitude at still lower elevation angles can be estimated.

Figure 1

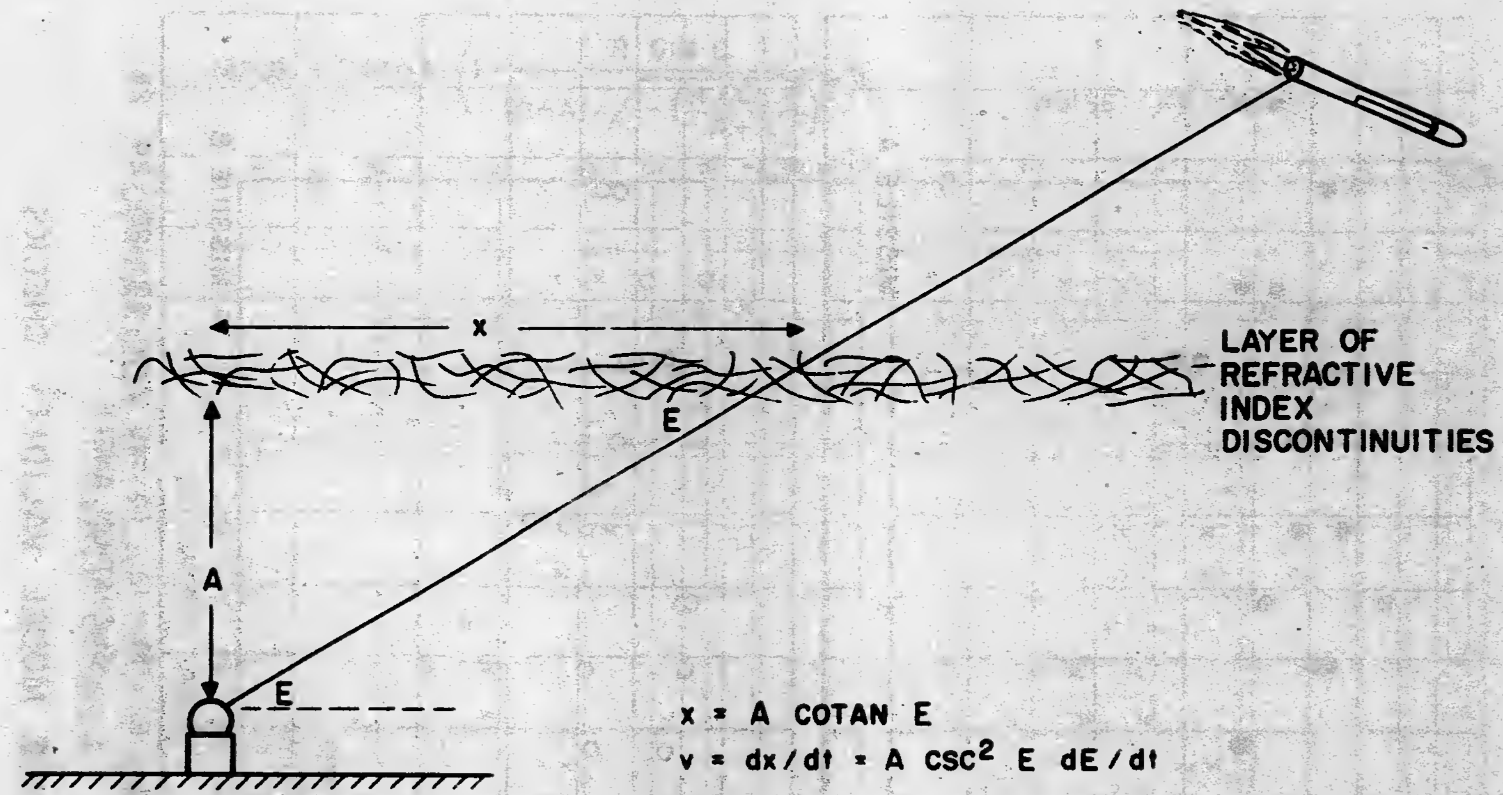


NOISE AS A FUNCTION OF ELEVATION ANGLE



ATMOSPHERIC NOISE CHARACTERISTICS

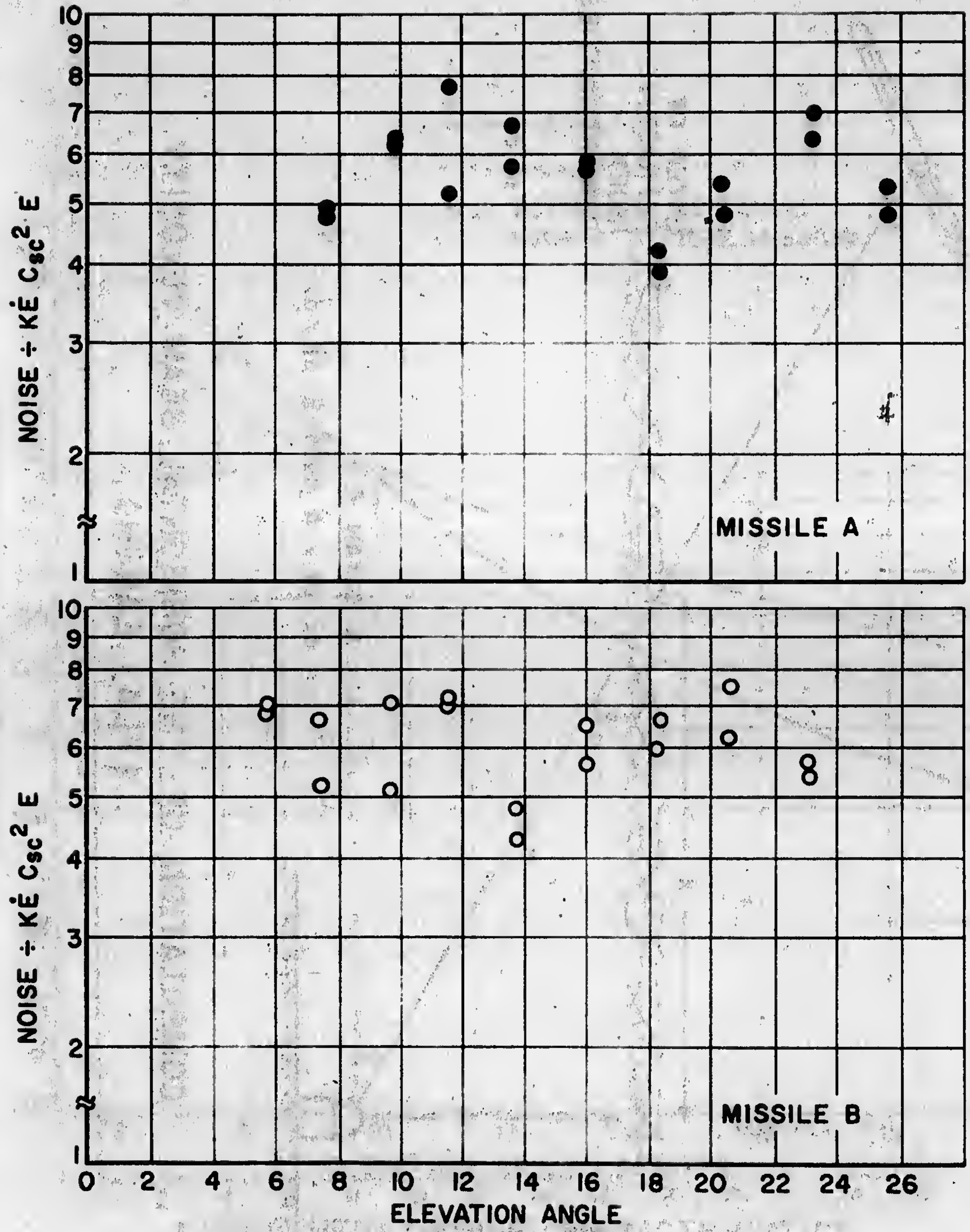
Figure 3



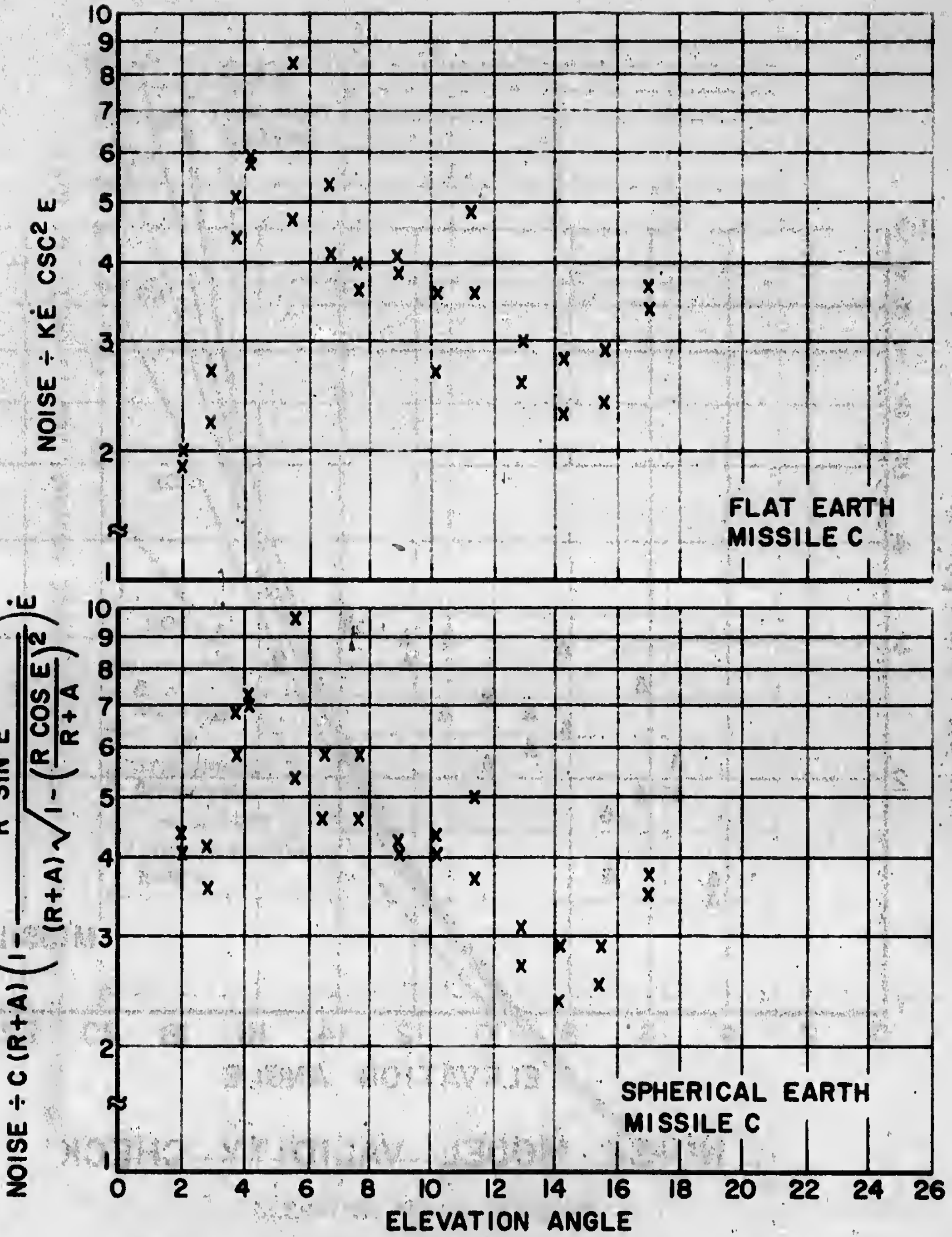
$x = A \text{ COTAN } E$

$v = dx/dt = A \csc^2 E \; dE/dt$

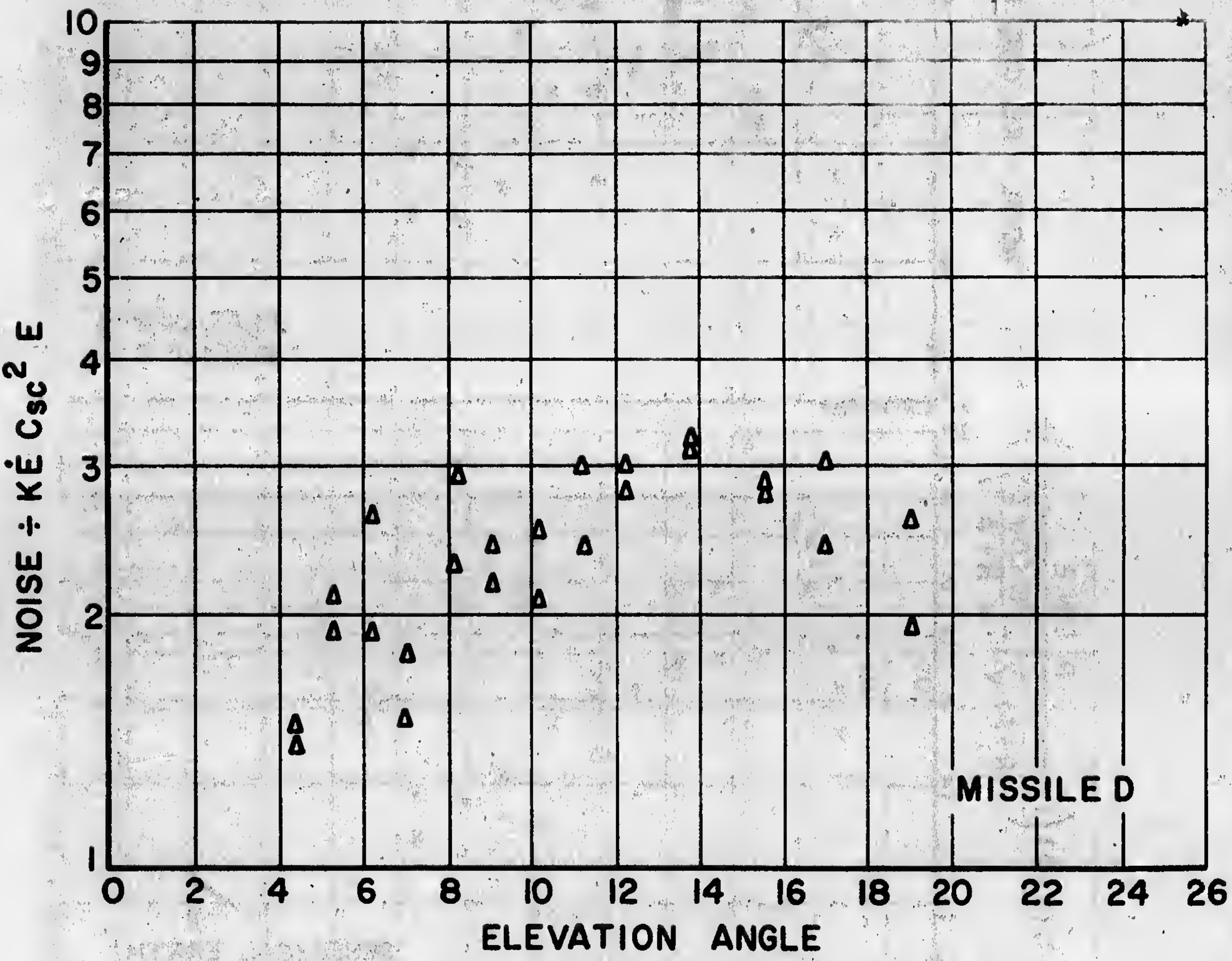
COMPUTATION OF THE HORIZONTAL BEAM VELOCITY
(FLAT EARTH)



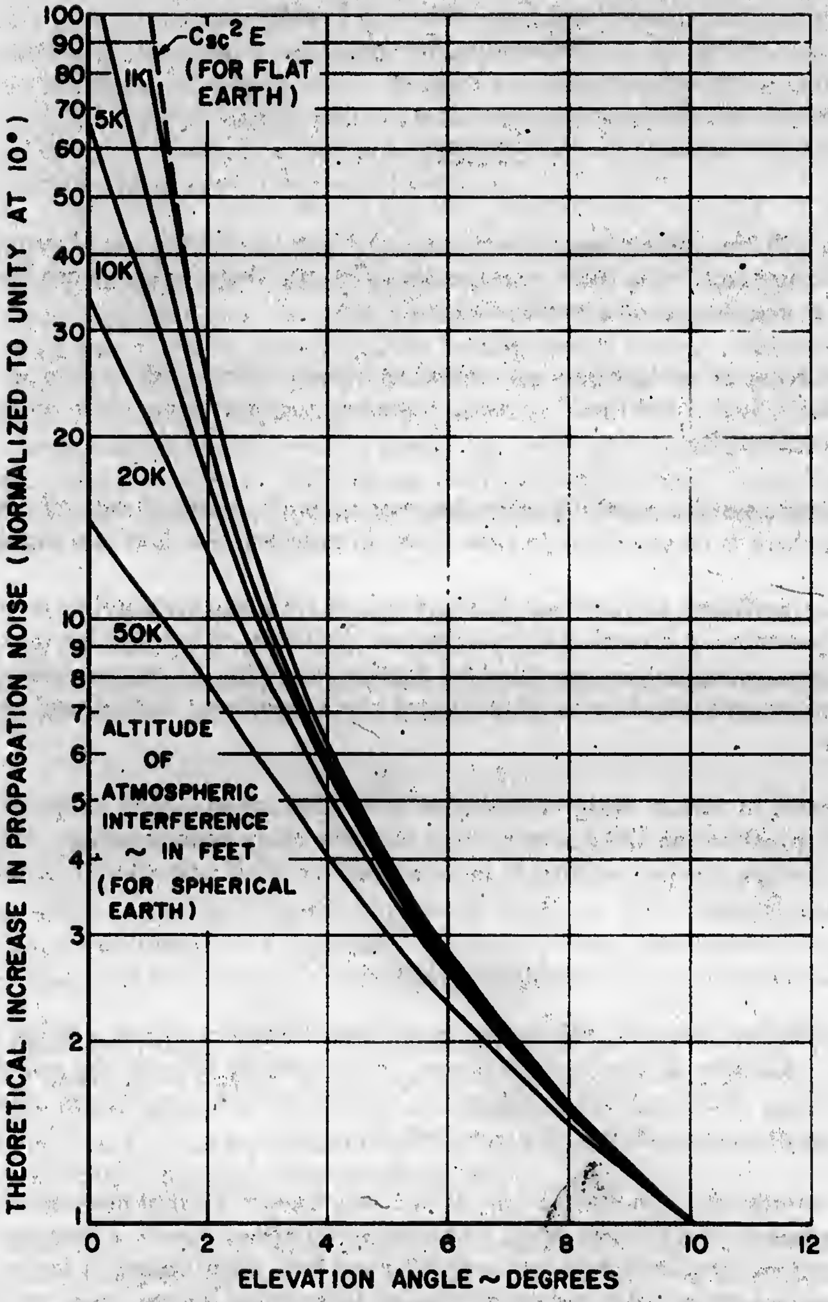
NOISE MODEL VALIDITY CHECKS



COMPARISON OF FLAT EARTH AND SPHERICAL EARTH MODELS



NOISE MODEL VALIDITY CHECK



THEORETICAL EFFECTS OF ELEVATION ANGLE AND ALTITUDE UPON NOISE MAGNITUDE

OPTICAL SYSTEMS FOR MEASURING POSITION IN SPACE

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Summary

There exist certain distinct advantages in accuracy and economy in the use of systems to measure vehicle position which operate at optical wavelengths. The positions measured are usually of critical importance to the success of a space mission.

There are several types of navigational and tracking systems which produce data in an immediately usable form. Such "real time" systems, however, sacrifice some relative accuracy to gain speed of data availability.

Stellar metric camera systems presently offer the most accurate means of determining position in space; however, there is an unavoidable time delay in reducing the data to a usable form.

There is room for significant improvement in the errors of presently used stellar metric cameras, particularly in the geometrical optical characteristics of the lenses in both design and fabrication. The usefulness of such cameras can be extended by daylight use with the proper designs and techniques. Lunar-based cameras should also offer several advantages over Earth-based systems, but only at a high price.

Recent developments in optical design techniques using high speed digital computers have brought far greater sophistication and economy than had previously been possible. Vastly improved new optical designs are just waiting to be executed for application to the problems of Measuring Positions in Space.

Introduction

One of the major factors effecting the success or failure of missions in space is the determination of position as a function of time. To navigate, it is necessary to know the position of the vehicle at a given time. From position as a function of time it is then possible to determine the velocity and acceleration components of the vehicle's travel.

When one contemplates navigation in space, it becomes imperative that measurements of position be made by beams of electromagnetic radiation using either active or passive systems. When one further contemplates navigation at some distance from other bodies in space, active electromagnetic systems such as radar and opdar become impractical due to power requirements. One finds also that the shorter the wavelength of radiation used in observation, the smaller the instrumentation can be for a given accuracy. This area of technology is what is referred to as diffraction effects in optics or beam or lobe pattern in microwave propagation.

What are some of the positions in space which are of interest? Let us first concern ourselves with the navigation of a vehicle from one planet to another or from a planet to a satellite of some planet. At least three different measurements are generally required to determine the position of the vehicle within the solar system. These measurements must include as a reference two sightings on bodies of known position within the solar system, such as the sun, a planet, or a satellite of one of the planets. The best solar system reference is usually the one nearest at hand. It is not our intention to discuss the geometry of the problem beyond what is necessary to identify the types of measurements to be made. Another class of positions of interest are geodetic positions of the surface features of various bodies and the actual trajectories of bodies influenced by the forces of these bodies. At least two applications of this latter class of positions are of importance: one is the calibration of various measuring instrumentation; another is the determination of geophysical characteristics of the major bodies of influence. These various applications have been described elsewhere.

Depending on the application, one may need to know the positional information almost immediately, i.e., in "real time", or one may be able to wait some reasonable length of time for documentary positional information. The question of,"is the data available in "real time"?", is a matter of degree; no data is usable at the instant that it occurs. However, there are generally two widely different classes of data when we consider the time between the event and the availability of the data concerning the event. The "real time" class of data system is limited primarily by electronic response time, whereas, the other class of data system is limited by human and mechanical response time in reducing the data to a usable form. Radar and opdar are examples of "real time" systems while ballistic and geodetic cameras are examples of "non-real time" systems.

The "real time" optical systems are generally required when it is desired or necessary to guide the vehicle on the basis of optical data. However, due to the usual sacrifices required in the precision of "real time" systems, photogrammetric techniques of documentation are usually used for precise calibration of real time optical and electronic position sensing systems.

Real Time Systems

Navigation Systems

There are many navigational optical systems in existence or under development today. There are several ways that these can be classed. We shall divide these systems into tracking and off-boresight classes, and this class we will divide into sequential and simultaneous sighting types.

Many systems exist which track on an object such as a star and readings of angle from some reference are made which assume that the system is pointing exactly at the object. There are a few systems, one of which we will describe in some detail, which are capable of reading the position of the object anywhere within a field of view (usually of a few degrees in size). Members of the latter class are off-boresight systems.

Most systems take a sighting on an object and measure the object's coordinates in some reference system, usually inertial. Sightings on various objects are taken successively and the relation of the object coordinate system to the coordinate system of the navigation system can be computed. Such a system must take its sightings sequentially and refer to a "fixed" coordinate system. Another type of device which can be used would have two sighting devices which simultaneously observe two objects and measure the angle between them. The angle between two objects is determined simultaneously but one must read the angle between other pairs of objects sequentially. If the system has three sighting devices and can measure the angle between each, three objects can be sighted simultaneously and position can be determined continuously, in "real time" so to speak. All the navigational systems except this last "TRISECTOR" method require a significant time lag for the successive sightings which are required and therefore are somewhat short of being truly "real-time" systems.

Dual-Field Space Sextant

We would like to describe here the optics of one navigational system which we have designed and built. The system is designed to be used in an off-boresight, sequential mode of operation. The instrument has been referred to as a "Dual-Field Space Sextant", but the term sextant is actually not as appropriate in this case as it would be for the simultaneous two sighting system described above. This system has been described elsewhere[1] in great detail. The first model (see figure 1) consisted of a telescope having a two-degree field of view and an optical system with a 165-degree, inside-out, quasi-stigmatic field-of-view, which were axially concentric and integrated to be mutually non-obscuring and confocal.

The wide field system was designed to look at planetary disks at close distances for determination of local vertical and stadiametric ranging. The field of view was turned "inside-out" to cause the image of the horizon of a planetary disk to shrink from the edge of the format toward the center as the planet is approached and the horizon subtends a larger angle. The black sky background appears in the center of the field. The "inside-out" field of view, therefore, allows the center of the receiver, a vidicon or orthocon television tube, which has higher resolution than the edges to be utilized when the stadiametric ranging and local vertical determinations are most critical at very close approach, to the planet. The utilization of the best resolution portion of the receiver at the most critical phases of navigation is the only motivation for the "inside-out" optical system.

The basic system chosen to satisfy the wide field requirements consists primarily of a toroidal element which resembles the bell of a horn. Figure 2 shows the wide-field system schematically. The horn element serves almost exclusively as a device to produce the desired inside-out field distortion and appropriate angular expansion of the annular field. The flat mirror, the negative achromat, and the positive achromat in the wide-field system serve only to transfer the image to the focal plane with appropriate magnification. The effective focal length of this total system is approximately three-tenths of an inch.

Let us consider what a field of view would appear to be when turned inside-out. Figure 3 shows diagramatically a potential field of view and what that field of view would appear to be when seen through the inside-out optical system. If one was to position this system with its axis pointed at the horizon, and observe a vehicle traveling down a highway in front of the instrument, the vehicle would appear to start from the center of the format as it appeared at the extreme edge of the field of view in actuality, and travel toward the edge of the format at which point it would disappear and reappear at the opposite edge, travel toward the center and disappear at the center as it actually disappeared from the field of view on the real horizon.

The narrow field of view optics is essentially a telescope with a small field of view. The receiver is a television vidicon tube or image orthocon. Functionally, the narrow field system of view is dictated by the usable diameter of the photo-active surface of the image tube and the number of elements or lines which can be resolved in this diameter. If, for example, an angular resolution of one second of arc is desired, the field of view when imaged on the photo-active surface of the image tube must not subtend more seconds of arc than the number of resolvable lines. The aperture's real diameter must also be large enough so that the limit of angular resolution imposed by diffraction is less than that imposed by the image-tube resolution. For a selected image tube and brightness of the dimmest star of interest, the effective aperture of the system necessary to give the desired signal-to-noise ratio can be determined. Consideration of the space sextant application requirements emphasises the desirability of the shortest folded system functionally practical.

The specific optical system chosen for the first model produced was that of the Schmdt-Cassegrain form of telescope. Figure 4 is a scale drawing of the combined optical systems including the Schmidt-Cassegrain System and the wide field optical system. The design of this system was carried out entirely, except for sliderule calculations, on the IBM 7090 Computer, and much of the optimization was done by a semi-automatic lens design program. Figure 5 shows the feasibility model of this design which was fabricated. Figure 6 shows a second instrument which has been produced with slightly different specifications such as: a one degree field of view at higher resolution; and an inside-out optical system with a smaller dead space in the center of the format. The second model has a functioning mechanical change-over system from one field of view to the other. One of the objectives of this design is to allow changing from one field of view to the other without the movement of any optical component. This feature should reduce any danger of misalignments in the course of field changeovers. In the second model, the change-over is accomplished by sliding a baffle tube down from the focal plane to obstruct the beam coming from the telescope primary mirror to the secondary mirror. As this tube reaches the end of its travel, it activates a shutter mechanism within the wide field system to open these optics and allow light from the wide field system to reach the focal plane while the narrow field system is obstructed. A simple one quarter turn of a crank shaft accomplishes this in practice. Figure 7 is a photograph of the second model produced which has an outside diameter of 6 inches. It will be noted in Figure 6 that the second model is not a Schmidt-Cassegrain System but more nearly resembles the Maksutov-Cassegrain System. In actuality, however, the system is neither a Schmidt nor a Maksutov System but intermediate between the two with a curved corrector as in the Maksutov but with heavy aspheric figuring as with the Schmidt corrector. We have, therefore, chosen to call this system a Maksutov-Schmidt Cassegrain design.

Real-Time Tracker

There exists a desire in the missile field and in some other applications for a real-time tracking optical system of the theodolite type. Photographic-recording theodolites have the disadvantage of the time required to reduce the data. Electronic or radar-type tracking systems have the disadvantage that the wave-lengths used are long and therefore the resolution is inherently low. One of the few existing real-time trackers with which we have some familiarity by virtue of designing the optical system may be considered to be equivalent to a passive radar system operating at very much shorter wavelengths or higher frequencies. The actual aperture of this system is 6 inches. The absolute tracking accuracy of this system is expected to be less than 10 seconds of arc. The resolution of this system on the other hand, is anticipated to be a few seconds of arc. The problem in the optical design of this system was not particularly one of image quality for the field of view is essentially a very few minutes of arc. The design problem here was one of trying to optimize the various opto-mechanical parameters of the system to reduce bore-sight drift to a minimum under the conditions of the thermal environment anticipated. The actual optical system was a simple Cassegrain telescope, but the relative magnifications used and the positions of the element were studied in great detail to minimize the bore-sight drift. The focal length of the system was 200 inches. The design study consisted primarily of considering the effects of tilts and decentrations of the various elements of the system with respect to their effect on the boresight of the system. In this case, the tilts and decentrations anticipated were not of a magnitude to significantly effect the image quality of the system. This characteristic is essentially similar to the stellar-geodetic cameras that we will describe subsequently. Real-time tracking theodolites of this type and other types have distinct advantages when data is required immediately. However, they contain inherent difficulties mechanically, electronically, and optically that cause a reduction in the precision and accuracy of these systems as compared to some of the photogrammetric methods to be described next.

Stellar Metric Cameras

Ballistic Cameras

The type of ballistic cameras to which we refer are essentially photogrammetric or aerial cameras pointed at the sky. In the most accurate application of such cameras, a background of stars is photographed and the vehicle of interest passes through the field of view emitting or reflecting light of its own or of a beacon attached to it, and the stars and the beacon are photographed with appropriate shuttering sequences to identify the times at which the stars and the vehicle were at given positions. The procedures used are essentially similar to the procedures of astrometry used for many years by astronomers determining the positions of various stars and various celestial bodies passing across a background of stars. Astronomers, however, have very long focal length instruments and ordinarily track the star background. In ballistic camera work, the cameras are fixed with respect to the earth on most applications and the trails of stars are recorded on the photographic plates with time breaks for identification of position as a function of time. Most of the early work with ballistic cameras was done with very short focal lengths of from 100 to 200 mm. This afforded a wide angular coverage when a standard photographic plate was used, and this had definite advantages in the work with short range missiles and aircraft. However, the severe limitations of positional accuracy due to the short focal length or small scale ratio resulting therefrom have made it desirable in many applications to have longer focal length systems. The Atlantic Missile Range at the present time has systems of 300, 600, and 1000 mm focal lengths for this reason. The Instrument Corporation of Florida is quite intimately familiar with this instrumentation since we have built the 1000 mm cameras and have also produced 600 mm and shorter focal length cameras.

Brown[2] has shown the error budget for the PC 1000 cameras as they exist today under normal conditions. Figure 8 shows a table of the various errors taken after the work of Brown with our own additions. The figures to note at this moment are that under normal conditions today the orientation of the camera is known with an error of less than one arc second and that under the improvements that are being made daily, it is anticipated that within a short time, approximately .6 arc seconds accuracy can be obtained. The determination of the direction of flashes from a beacon, such as the ANNA satellite, can be made to essentially equivalent accuracy. Figure 9 and Figure 10 show an operating 1000 mm Camera System inside a dome with a radial louver shutter. This particular shutter was developed by the Instrument Corporation of Florida and is capable of a chopping sequence at a maximum rate of 55 chops per second with a transfer time from close to open or vice versa of 5 milliseconds and a shutter position pick-off for precise measurement of the actual times of the opening and closing. The electronics associated with the systems for timing and shutter actuation are not shown in the figures.

Geodetic Cameras

Ballistic cameras can be applied to the measurement of position of orbiting satellites or flares exploded at high altitudes. If a network of three or more cameras at known positions on the surface of the earth can determine the accurate positions with respect to the network of the flares or satellites at a given time, cameras at other sites whose positions are not accurately known can be accurately located by resection. This is essentially the same procedure used at the known sites to locate the satellite except that in the latter case we know the position of the satellite or flares and we determine the position of the camera by photographing the satellite against a star background. These procedures can determine with extreme accuracy the positions of points on the surface of the earth. When sufficient numbers of accurate positions have been determined, it then becomes possible to determine the precise orbits of various bodies, satellites, etc., that encircle the earth. From the perturbations of these orbits, it is possible to determine various other geodetic data such as mass distribution of the earth, atmosphere density at high altitudes, etc.

Future Improvements of Geodetic and Ballistic Cameras

Referring to the table in Figure 8, we have indicated the possible improvements which can be expected with an earth-based 1000 mm focal length camera in the near future. One major improvement being worked on at the present time is the reduction of star catalog errors. Another error which is being attached from both the analytical and the fabrication point of view is the residual tangential distortion of lens systems. Most of the other errors contributing to the error budget can be reduced by small amounts through improved equipment and techniques.

We also show in Figure 8 the anticipated error budget from a 2000 mm land-based geodetic camera and this is expected to have about one half the error of a 1000 mm camera.

Another area of future improvement in the use of geodetic and ballistic cameras is being pursued at the present time and this is daytime usage of stellar geodetic cameras. It can be shown that the relative merit of various cameras with respect to producing high contrast images of point sources against a continuous sky background is directly proportional to the square of the focal length. It is at first surprising that the contrast is not also a function of the aperture. This indicates then that a 1000 mm camera would have relative contrast eleven times greater than a 300 mm camera when photographing the same object under the same conditions. Because of the fourth power law of scattering in the blue sky, it is fairly obvious that it is necessary to work in the longest wave length region possible, and that would be the red or near infrared region of the spectrum. The limitations placed on the focal length which can be used are primarily due to the field of view desired and the format which can be used, the standard ballistic camera plate format at the present time being about 180 mm square. This gives, for example, a field of view of 10° by 10° at 1000 mm focal length. At 2000 mm, this would be cut to about 5° by 5° square. Various daylight systems currently are under study including laser source systems and other optical beacons to be placed on the vehicle to be tracked.

Another vast new area in the field of geodetic cameras may be opened up by taking cameras outside of the earth's shimmering sea of atmosphere. In determining the direction of flashes from a strobe satellite such as ANNA, the largest error is introduced by the high frequency atmospheric shimmer as we have shown in Figure 8. Another error contributed by the atmosphere is residual paralactic refraction error. Simply by removing these two factors, it is possible to improve the accuracy and determination of the flashes on the order of 60%. Lunar based geodetic cameras or ballistic cameras may shown great potential because of the unlimited weather conditions, perpetual night sky, and slow diurnal rate. It is not inconceivable that such cameras would be useful in the guidance of vehicles in inter-planetary space because of their extended range of usefulness and the very high angular resolution obtainable. Such systems will probably be of great focal length and approach very closely the more classical application of astrometry. Lunar-based cameras, however, are confronted with several rather formidable problems such as radiation, heat, cold, and high energy particles which would tend to expose the plate unless very careful shielding is done. There would have to be a rather complete lunar-based timing and communication system to determine time from astonomical constants. The photographic plates would have to be maintained at some temperature at which they are useful. The cameras would have to be protected from the effects of meteorites and there is apt to be operator's problems in that the clumsiness of probable protective clothing for the human will not allow precise work and he would have trouble viewing through conventional finder telescopes. Complete plate development and plate reduction facilities will have to exist on the moon for the data to be practically useful. It is clear that automation would have to be used extensively to make a lunar geodetic camera station functional.

Stellar geodetic cameras would be quite useful in determining details of the lunar surface positions, shape, and mass distribution as they would be determined on the earth. In fact, photogrammetry and stereo photogrammetry would seem to be the only reasonable first approach to a general survey of the moon. If the moon were 1/4 the diameter of the earth, the surface therefore would be only 1/16 that of the earth. However, photogrammetrists only have 1/4 of the earth's surface in land to be measured. This means that the lunar surface consists of about 1/4 the land area of the earth and photogrammetrists know how many man-hours it has taken to arrive at our present state of geographic knowledge of the earth, and they further know how slow the rate of progress was before the advent of photogrammetry when only the conventional theodolite survey was performed.

Lunar photogrammetry is a vast new world which has only become of interest to many in the past decade. The most thoroughly distributed publication in the field today is probably the Air Force's Lunar Chart series available from the Government Printing Office. The scale is the same as that of the World Aeronautical Charts and the price is only twice that of a WAC.

Optical Design

The instruments used in photogrammetry including stellar geodetic and ballistic cameras depend very heavily on the quality of the optical design of the system and its execution by fabrication. There have been many advances in the past decade in the field of optical design which should allow considerable improvement in the various photogrammetric instruments. However, very limited use has been made, to date, of these advances.

One area of advance has been promoted by the advent of the high-speed, high-capacity digital computer. The pioneering work of Dr. James Baker in the era containing the Second World War would have made unbelievable strides if performed on the computers available today. The computers of today offer tremendous economy in both time and money to the design of optical systems. Whereas relatively simple photographic lenses would take years to design in the past, much more complex designs can now be achieved in months or even weeks. Semi-automatic design programs have been developed which take almost all of the burden from the designer and allow him to think only the most penetrating thoughts required for the optical design and not be bothered with mundane and tiresome tasks. Although all these programs are still in the very early stages of development, they are already proving to be invaluable tools, and the organization which attempts to pursue this business without such capabilities is sure to find that its designs are not competitive in either price or quality. Another advantage of the computer era of lens design is that much more thorough work can be done in the optimization of designs and many more factors can be practically included in the design rather than considered by only a cursory inspection.

Another major area of improvement is that of optical materials both for refraction as in glasses and other transparent media and reflection as in metallic mirrors and other types of mirrors. One should also comment on the advances in coating technology for high reflection, anti-reflection, and spectrally selective coatings for filters. Not only have there been improvements in the available materials and material processing techniques, but also there have been very significant improvements in the methods of selecting appropriate materials for color correction. Hertzberger[3] and others[4] have made great strides in developing the selection techniques. It has been shown that using only refracting elements it is possible to design lenses which would not be detectably different from mirrors with respect to color correction. Color correction has been one of the most serious faults of stellar geodetic systems. We have done extensive theoretical and experimental investigations in this area recently to show the extent to which these difficulties exist. The useful spectral range of even the best stellar geodetic systems is only from 1000 to 1500 Angstroms wide. However, the sensitivity of the plates used is more than 3000 Angstroms wide and in the night operation of stellar geodetic cameras there is no difficulty with the blue sky background. It should therefore be possible to use all of the visible spectrum and perhaps some of the radiation in the near ultra-violet and infrared to advantage. If this were accomplished, a given aperture would collect on the order of two to three times as much usable energy and therefore be more than one f-stop faster than present systems at the same aperture. This is definitely one of the fertile areas which should be pursued next in the field of stellar geodetic cameras.

The new techniques have, of course, had considerable advantage in the development of all types of optical systems. Navigational systems and real-time tracking systems have benefited from these new techniques somewhat already. The two dual field star trackers described earlier were designed exclusively on high-speed digital computers and much of the optimization was done by semi-automatic lens design programs. The two systems described also included 4 and 2 aspheric surfaces respectively. Such surfaces are very difficult to design and evaluate by other than digital computation[5]. The two systems described were designed in their entirety, including the telescopic and inside-out optical system, in less than one hour of computer time per system. The time between conception of the system and the final detailed working design was only a few weeks, although we must admit that these were designed on an "as soon as possible" basis.

The techniques and tools of the optical design field and the optical industry in general are ready to provide the instrumentation necessary for all of the many far-reaching programs in the measurement of positions in space, such as navigation, tracking, photogrammetry, and geodesy.

Footnotes

1. H. E. Brunelle and R. R. Willey, "Coaxial, Dual-Field Optics for a Space Sextant", Applied Optics, 2, 1265-9 (1963)

2. D. C. Brown, N. Bush, J. L. Sibol, "Study of the Feasibility of Rocket and Satellite Approaches to the Calibration of Tracking Systems", D. Brown Associates, Inc. (1963)

3. M. Herzberger, Modern Geometrical Optics, Interscience (1958)
also Herzberger and N. R. McClure, "The Design of Superachromatic Lenses", Applied Optics, 2, 553 (1963)

4. R. R. Willey, "Machine-Aided Selection of Optical Glasses for Two-Element, Three-Color Achromats", Applied Optics, 1, 308 (1962)

5. R. R. Willey, "Machine-Aided Design of Lenses Containing More than One Aspheric Surface", J. Opt. Soc. Am., 52, 608 (1962) (abstract of presented paper).

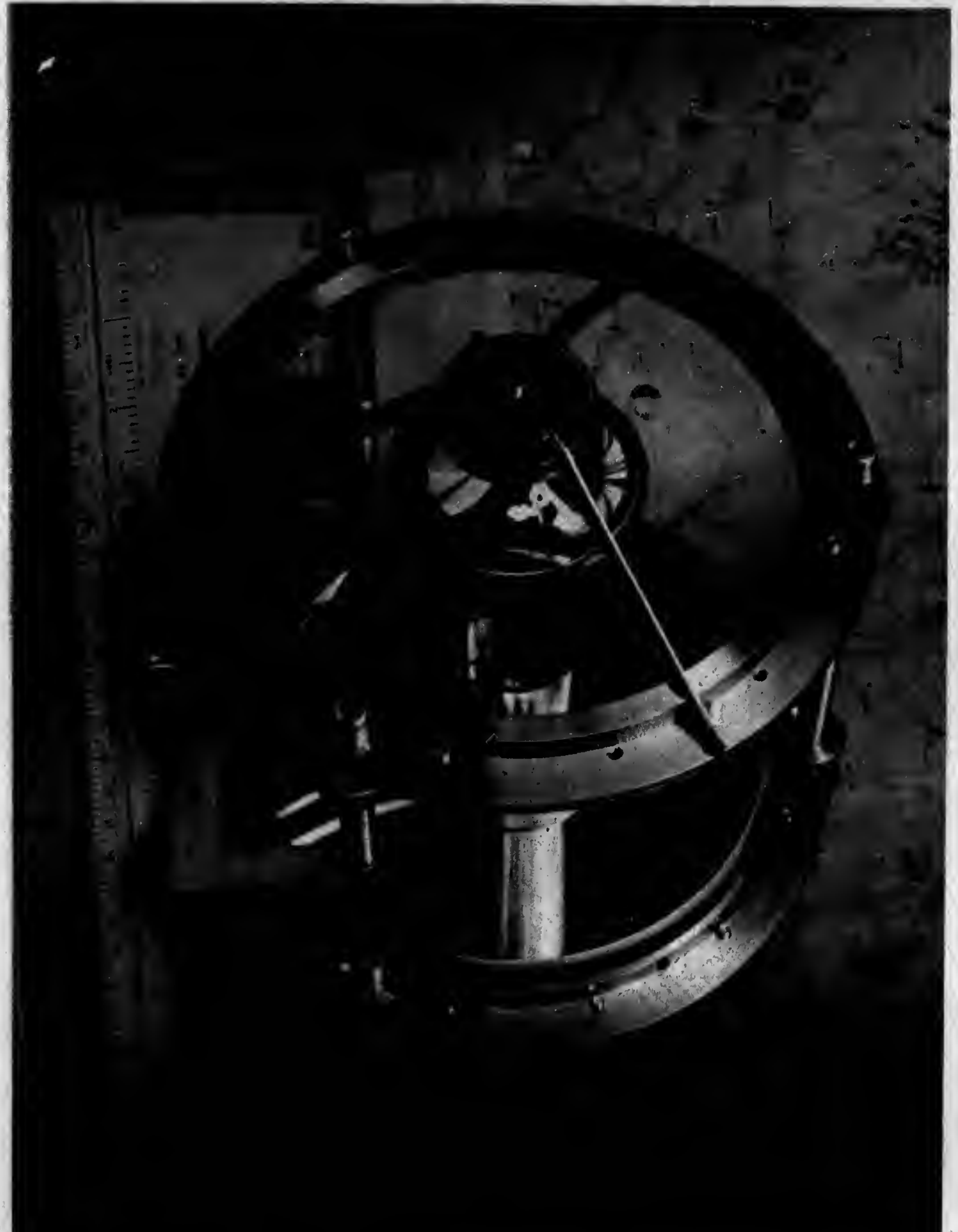

Figure 1. DUAL FIELD SPACE SEXTANT

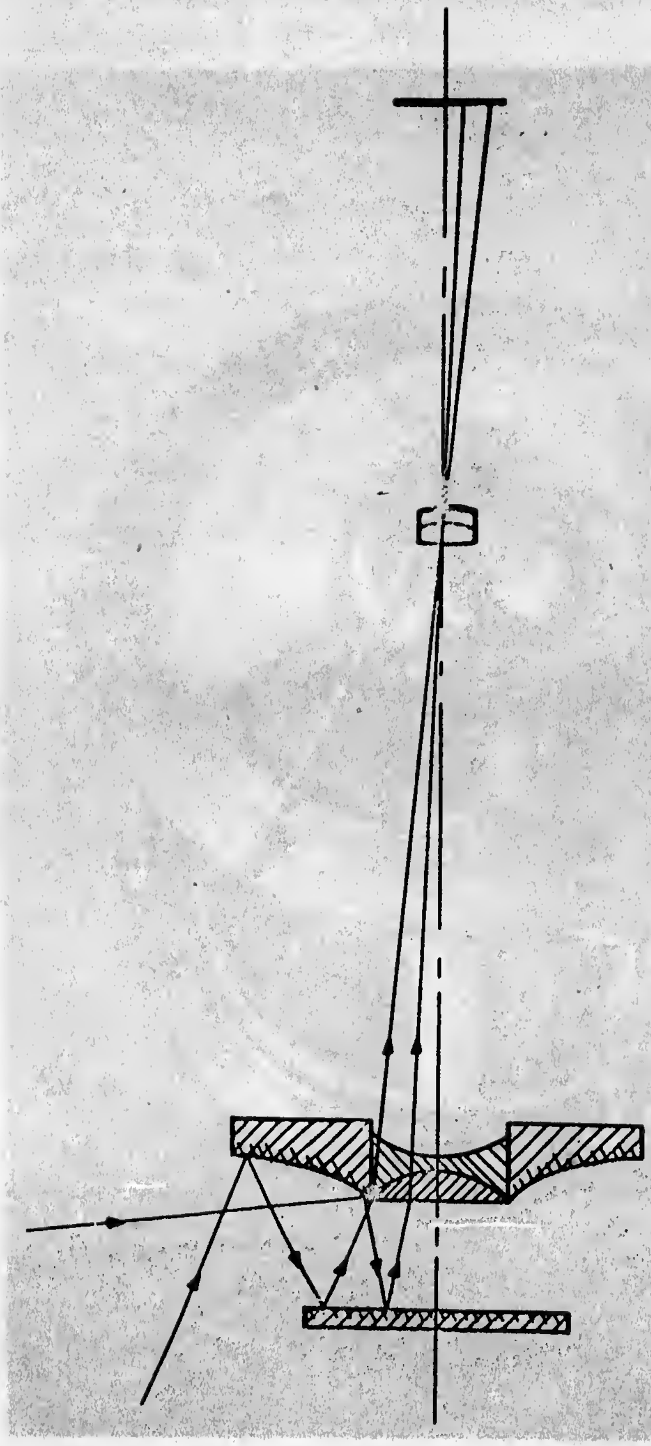

Figure 2. Annular Wide Field System

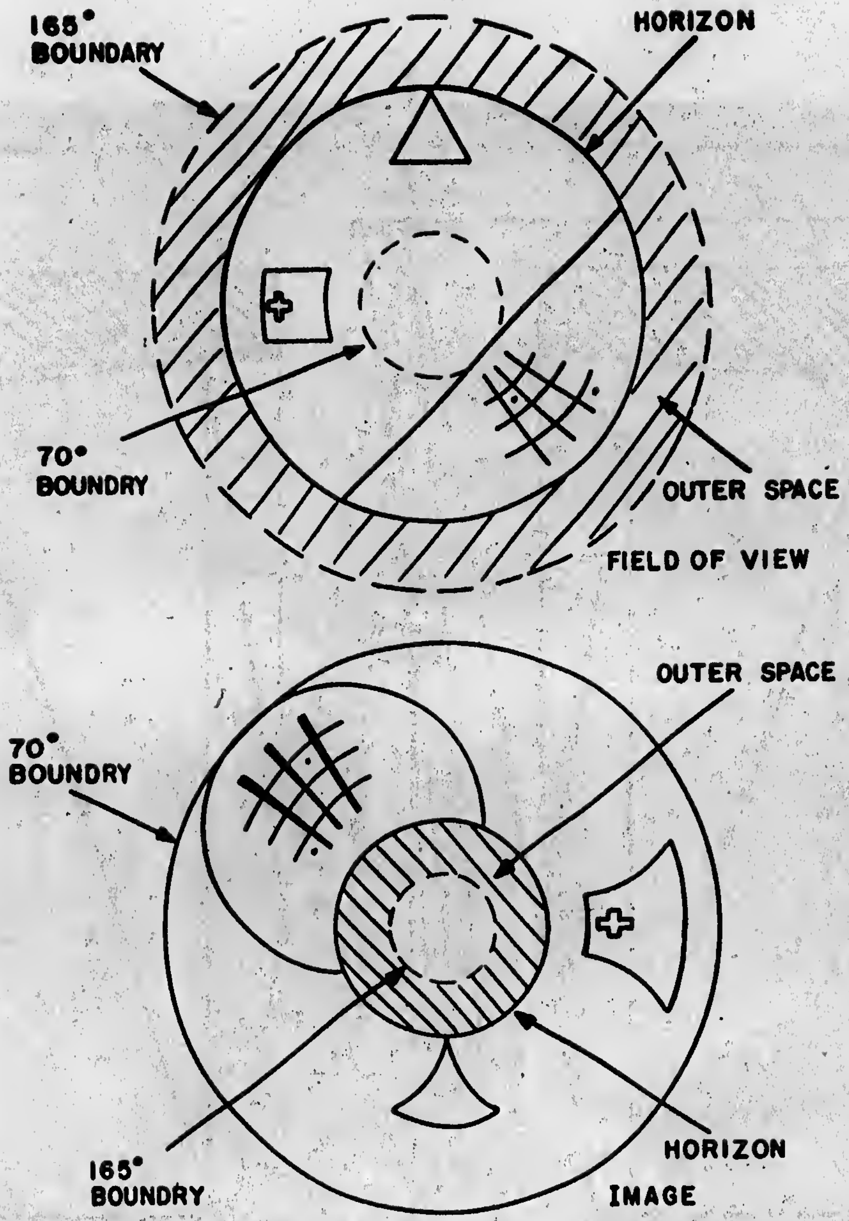

Figure 3. Space Sextant Monitor Display of Field of View as seen by Image Tube Through the Wide Field System

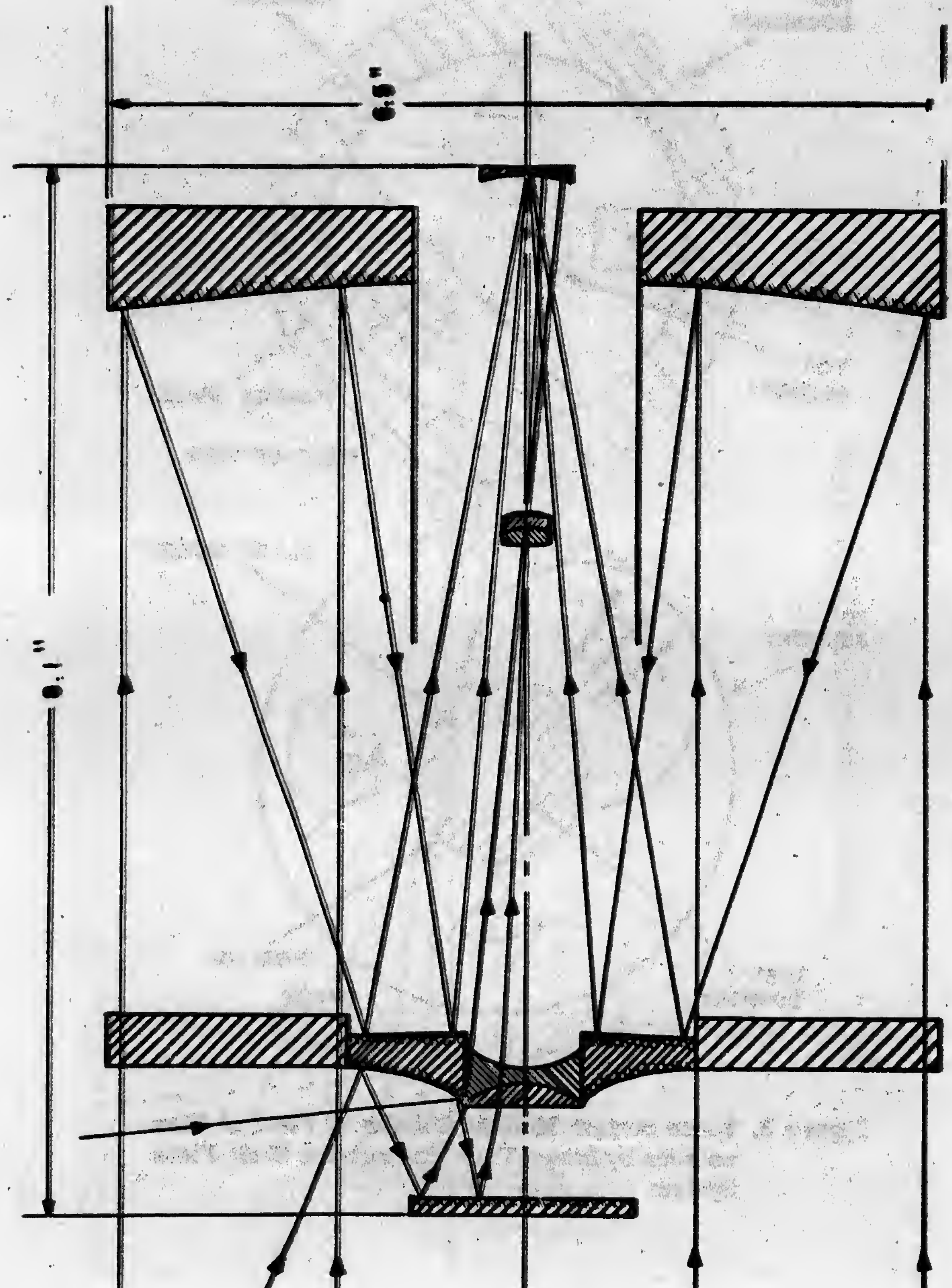

Figure 4. Combined Optical System

Figure 5. COMBINED OPTICAL SYSTEM ASSEMBLY

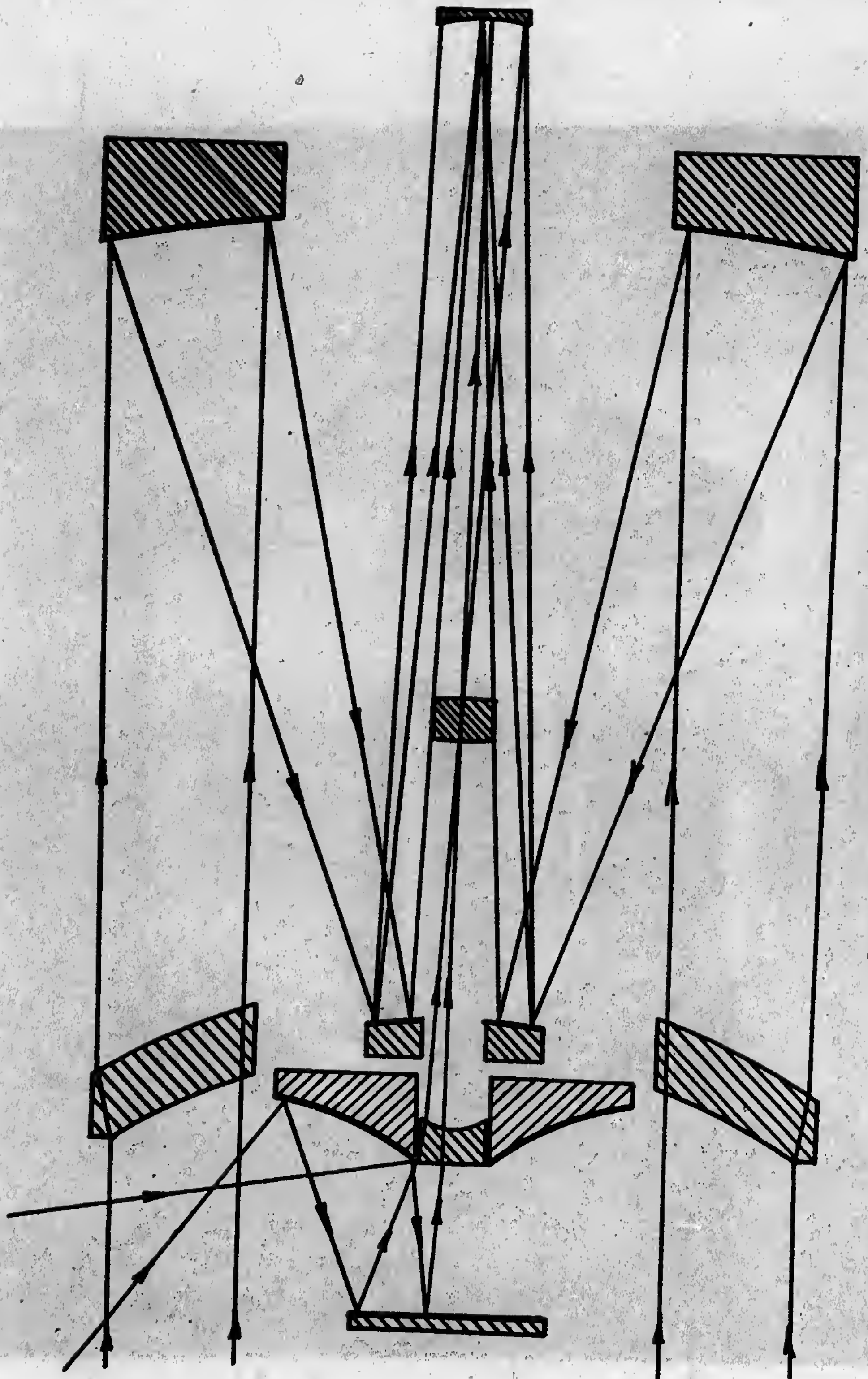

Figure 6. Second Model Optical System

Figure 7. SECOND MODEL ASSEMBLY

Geodetic Cameras
RMS Error Contributions
Under "Normal" Conditions

	ERRORS IN MICRONS				
Calibration of Orientation:	Earthbased 1000 mm today	Earthbased 1000 mm tomorrow	Lunar Based 1000 mm tomorrow	Earthbased 2000 mm tomorrow	Lunar Based 2000 mm tomorrow
1. Catalog Errors	3.0	1.0	1.0	2.0	2.0
2. Residual Tangential	2.0	1.0	1.0	1.0	1.0
3. Random setting error	1.5	1.0	1.0	1.0	1.0
4. Emulsion Instability	1.5	1.0	1.0	1.0	1.0
5. Camera Instability	1.5	1.0	0.5	1.0	1.0
6. Low Frequency Atomospheric Shimmer	1.0	1.0	0.0	0.5	0.0
7. Residual Radial Distortion	1.0	1.0	1.0	1.0	1.0
8. Residual Comparator Errors	1.0	0.5	0.5	0.5	0.5
9. Timing Errors	1.0	0.5	0.5	1.0	1.0
10. Residual Differential Refraction	0.5	0.5	0.0	1.0	0.0
Pooled RSS Totals	4.9 (1 sec)	2.8 (.6 sec)	2.4 (.5 sec)	3.5 (. 3 sec)	3.2 (.3 sec)
Determination of Direction of Flashes					
1. High Frequency Atomospheric Shimmer	2.5	2.5	0	1.5	0
2. Residual Tangential Distortion	2.0	1.0	1.0	1.0	1.0
3. Random setting errors	1.5	1.0	1.0	1.0	1.0
4. Emulsion Instability	1.5	1.0	1.0	1.0	1.0
5. Residual Radial	1.5	1.0	1.0	1.0	1.0
6. Residual error in Calibration of Orient.	1.0	0.5	0.5	1.0	0.5
7. Residual Paralactic Refraction	1.0	1.0	0	1.0	0
8. Residual Comparator Errors	1.0	0.5	0.5	0.5	0.5
Pooled RSS Totals	4.5 (.9 sec)	3.4 (.7 sec)	2.1 (.4 sec)	2.9 (.29 sec)	2.1 (.2 sec)

Figure 9. 1000 mm GEODETIC CAMERA SYSTEM WITH SHUTTER AND DOME AS SEEN IN USE

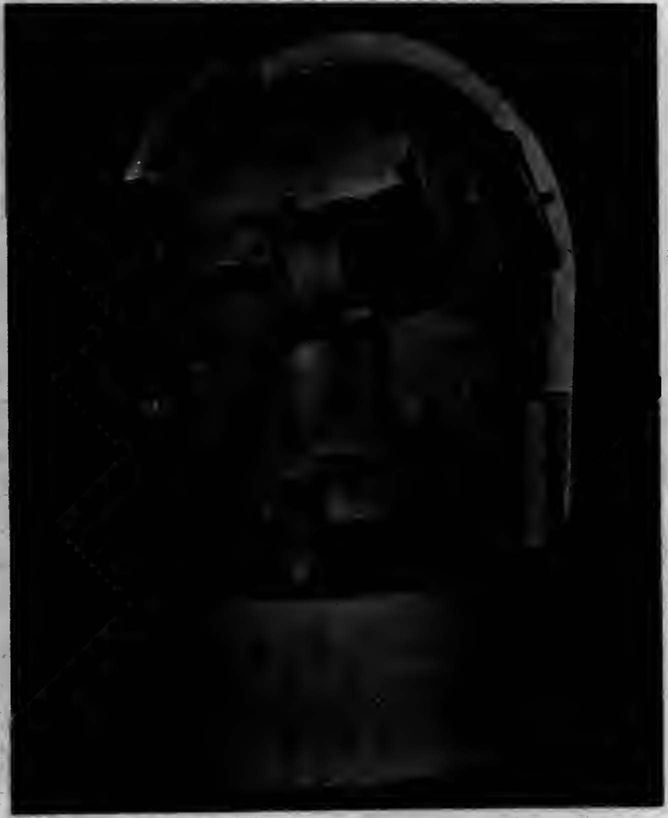

Figure 10. 1000 mm GEODETIC CAMERA SYSTEM WITH SHUTTER AND DOME, CUTAWAY VIEW

Preliminary Report

PHOTOGRAMMETRIC OCEAN SURVEY EQUIPMENT (POSE) PROGRAM

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ABSTRACT

The accurate geodetic position of underwater hydrophones and transponders must be known to determine the precise location of missile impacts on the Atlantic Missile Range. Accurately located transponders will provide location data for missile guidance and accuracy evaluation as well as exact positioning of instrumented ships used for missile, satellite and space vehicle tracking. The geodetic position of many Missile Impact Location System (MILS) hydrophones contains a sizeable error. These hydrophones can be accurately located with shipboard sonar equipment if the ship can be precisely positioned geodetically. Since no other system was available for accurate ship positioning, the Photogrammetric Ocean Survey Equipment (POSE) program was initiated.

Testing of photogrammetric ocean survey equipment has been performed at the AMR in cooperation with the U. S. Navy Oceanographic Office. The testing determined with what accuracy a ship could be positioned at sea. A stellar oriented camera on a stable platform was mounted on a roll and pitch motion simulator at Cape Kennedy and later on a ship at sea. This gyro-stabilized camera, with five land-based stellar-oriented cameras, observed an aircraft-borne strobe light against a stellar background.

Test results were used to compute the geodetic position of the gyro-stabilized camera. It was positioned to 1.22 feet Circular Probable Error (CPE) when mounted on the simulator. Later the ship's position was established to an accuracy of 2.56 feet CPE. The latter accuracy is comparable to positioning a ship at sea to 300 feet (CPE) while observing a satellite with position uncertainty of plus/minus 50 to plus/minus 100 feet at a slant range of 1000 m. Accuracy of the results greatly exceeded that expected with the equipment used.

To obtain ship's position with the gyro-stabilized camera, four types of target vehicles may be observed: aircraft, missiles, active satellites, and passive satellites. The target used depends on the distance of the ship from land, the observation method, and the equipment used. All offer high accuracy potential when properly observed.

INTRODUCTION

The Atlantic Missile Range (AMR) must provide the geodetic position of missile impacts for guidance and re-entry evaluation testing. Impact location can be determined relative to underwater hydrophones or transponders by time correlated distance measurements. The true geodetic position of these hydrophones or transponders must be applied to this relative data to obtain the geodetic position of impact.

The geodetic position of many of the hydrophones and transponders used at the AMR are greatly in error. Consequently, geodetic position impact determinations are also greatly in error.

Initially, the geodetic position of a calibration ship is needed to determine the geodetic position of the hydrophones or transponders. This is done by applying the geodetic position of the ship to the ship's position relative to the hydrophones or transponders. This can be found quite accurately by measuring the difference in time required for a sound wave to travel to several different hydrophones or transponders. Accurate geodetic positioning of the ship is a more difficult problem, especially when the ship is far at sea.

Accurately positioned hydrophones and transponders could also be used for geodetic positioning of instrumented ships which track missiles, satellites or space vehicles. They would also serve as reference points to calibrate navigation equipment and could be used in conjunction with ocean bottom surveys. If extended across oceans, they would permit intracontinental geodetic ties.

Ships in broad ocean areas can not be adequately positioned to meet AMR needs by such navigational systems as doppler, Loran or inertial navigators aided by star tracking. These systems satisfy ship navigation standards, but they do not meet the more stringent needs of geodesy.

The Photogrammetric Ocean Survey Equipment (POSE) program was initiated to help solve this problem. Stellar oriented cameras have long been used at the AMR as the accuracy standard for calibration of other instrumentation systems. Recently, these cameras have been used for accurate geodetic positioning of land-based tracking instruments. Mr. Duane Brown, while with RCA at the AMR, hypothesized that one of these cameras could serve equally well on a ship if the effect of ship's motion could be overcome. Mr. R. Sterling and Mr. N. Vance of the U. S. Navy Oceanographic Office (USNOO) became interested in the technique, as did the author. The USNOO obtained a surplus two-axis gyrostabilized platform, formerly intended for aerial photography. The platform was modified to accomodate a K-37 aerial camera and experiments were made with the mount at the Pacific Missile Range. Mr. R. Drake, Project Officer for Missile Impact Location Systems at the AMR became interested in this effort and suggested a joint project, using the USNOO platform and AMR ballistic camera equipment and facilities for conducting controlled equipment evaluation tests. The purpose of this report is to describe the testing of this equipment, the results obtained and future efforts planned.

DATE	PLATE #	TIME	INDUCED DRIFT (DEG. PER SEC.)	INDUCED DRIFT PITCH	INDUCED DRIFT ROLL	SIMULATOR MOTION (CPS)	SIMULATOR AMP. PITCH (DEGREES)	SIMULATOR AMP. ROLL (DEGREES)	CAMERA AZ	CAMERA EL	EXPOSURE (SEC)	INTERVAL (SEC)	STOP	LENGTH OF CAMERA RUN (MIN)
9-4	1	2025	.014	X		.05	X	X	340	70	.5	4	4.2	3.0
	2	2035	.014		X	.05	X	X	340	70	.5	4	4.2	3.0
	3	2045	.028		X	.05	X	X	340	70	.5	4	4.2	2.5
	4	2050	.028	X		.05	X	X	340	70	.5	4	4.2	2.5
	5	2056	.02	X		.05	X	X	340	70	.5	4	4.2	2.5
	6	2103	.02		X	.05	X	X	340	70	.5	4	4.2	2.5
	7	2112	.02	X		.05		X	340	70	.5	4	4.2	2.5
	8	2121	.02		X	.05	X		340	70	.5	4	4.2	2.5
	9	2135	.028		X	.05	X	X	340	70	3.5	4	4.2	2.5
	10	2143	.028	X		.05	X	X	340	70	3.5	4	4.2	2.5
9-5	1		.014	X		.05	X	X	199	69	.5	4	4.2	3.0
	2		.014	X		.05	X	X	199	69	.5	3	4.2	3.0
	3		.014		X	.05	X	X	199	69	.5	3	4.2	3.0
	4		.014		X	.05	X	X	199	69	.5	3	4.2	3.0
	5		.014	X		.05	X	X	199	69	.5	3	4.2	3.0
9-10	1		.018	X		.05	±4	±2.5	350	69	.2	4	4.2	3.0
	2		.018		X	.05	4	2.5	350	69	.2	4	4.2	3.0
	3		.018	X		.05	4	2.5	350	69	.3	4	4.2	3.0
	4		.018		X	.05	4	2.5	350	69	.3	4	4.2	3.0
	5		.018	X		.05	4	4	350	69	.4	4	4.2	3.0
	6		.018		X	.05	4	4	350	69	.4	4	4.2	3.0
	7		.028	X		.05	4	4	350	69	.4	4	4.2	2.5
	8		.028		X	.05	4	4	350	69	.4	4	4.2	2.5
9-11	1		.018	X		.05	4	4	190	71	.2	4	4.2	2.5
	2		.018		X	.05	4	4	190	71	.2	4	4.2	2.5

TABLE 1
GYRO-CAMERA
PERFORMANCE TEST DATA

TEST EQUIPMENT

The decision was made to conduct joint tests using the USNOO stable platform and the AMR camera equipment. The Navy platform, built by Aeroflex Laboratories Inc. and designated ART-13, is a two-axis, one gyro mount permitting stabilization in roll and pitch, but not in azimuth. The platform is limited to roll and pitch amplitudes of plus/minus 7°. Stability of the platform is reported to be 1 to 2 arc minutes. It is stabilized by torquers, responding to signals from the gyroscope sensors. The platform has been modified to permit plus/minus 3° induced drift in both roll and pitch. The reason for this is to permit a gradual rotation of the camera optic axis during operations, resulting in spreading the stellar images across the photographic plate as a continuous trace. Induced drift rates varying from 0.01 to 0.10 degrees per second can be used.

The AMR made available a Wild BC-4, 210 mm focal length, F 4.2 camera for use with the ART-13 platform. An adapter was made for mounting the BC-4 camera on the Navy platform and the entire assembly was balanced. The camera can be rotated 360° in azimuth and will elevate from 0° to 90°. A driver unit operates the camera shutter with a preprogrammed, time correlated tone signal.

The camera - stable platform combination is attached to a low vibration frame easily installed on a ship.

TEST METHODS

A Phase I - Performance Tests

Phase I consisted of hardware performance tests. The gyro stabilized camera was installed on a roll and pitch simulator, rented from Aeroflex Laboratories Inc. The simulator was caused to roll and pitch at varying amplitudes up to plus/minus 4°. Roll and pitch frequencies were varied up to 0.8° per second.

There was no provision for yaw motions. The stable platform compensated for the simulator motions, and in addition, was torqued in either roll or pitch to induce a drift that would cause the camera to sweep. Drift rates varied from 0.01 to 0.05° per second; camera exposure times varied from 0.2 to 3.5 seconds; exposure intervals varied from 1 to 5 seconds. The full F 4.2 aperture was used for each exposure. A tabulation of operating conditions during certain Phase I tests is shown in Table 1.

The objective of Phase I testing was: 1) to assure proper physical operation of the gyro-stabilized camera, and 2) to determine the best combination of camera drift rate, drift direction, exposure time and exposure interval that would produce optimum star images on the photographic plate. A 0.028° per second drift rate when combined with a 0.4 second exposure time and a 3 or 4 second exposure interval produced best results with the BC-4, 210 mm focal length camera at full aperture. Discrete, circular images varying from 40 to 120 microns in diameter were formed during the shutter-open period, depending on the magnitude of the star recorded. The 4-second interval between exposures prevented image overlapping. The direction of induced drift was not a critical factor for image recording. However, accurate balancing of the system is essential.

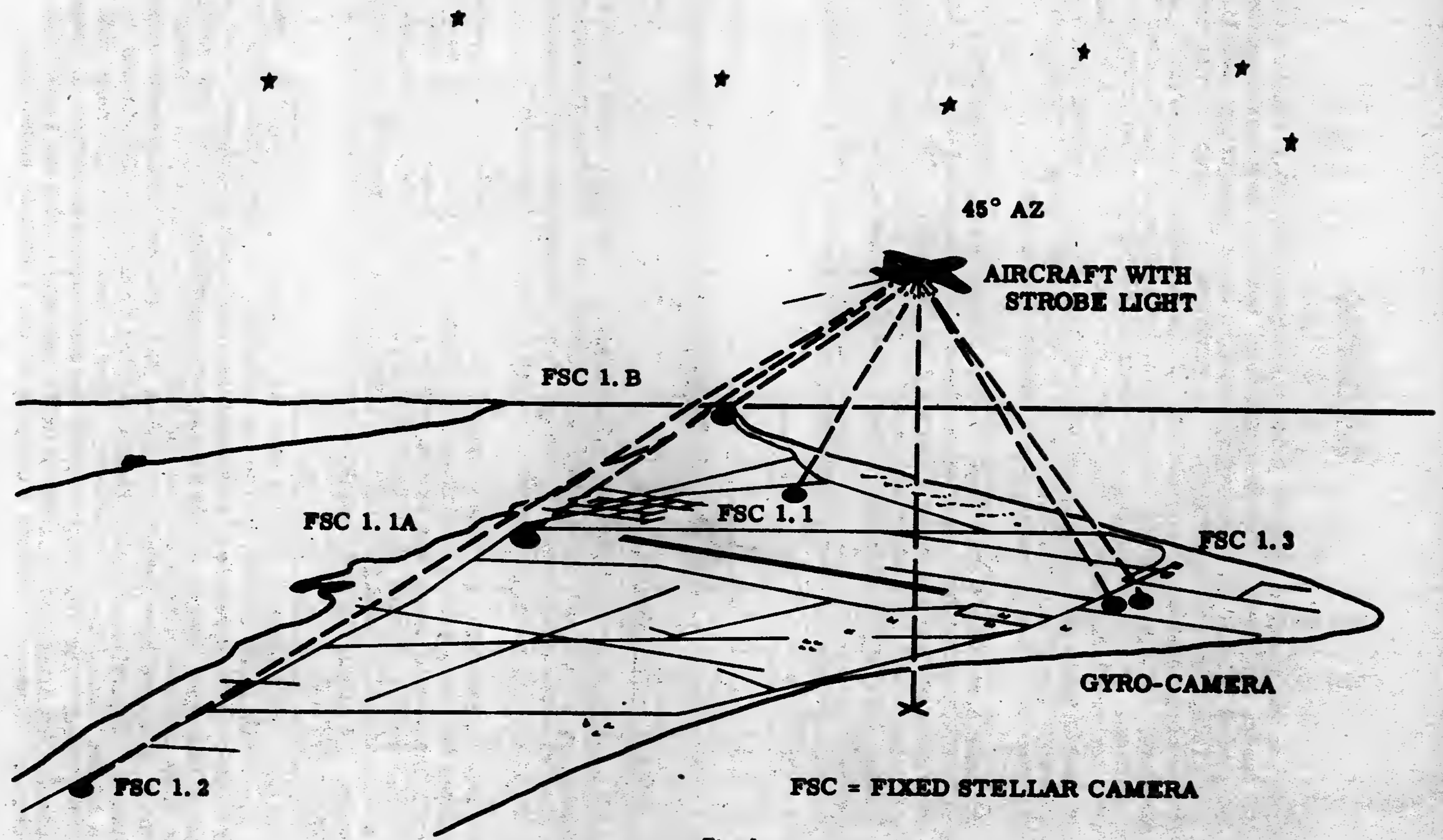

Fig. 1
POSE SIMULATOR TEST CONFIGURATION

Drift rate, exposure time, exposure interval and aperture must be closely controlled to obtain maximum data on the photographic plate. If the drift rate is too great the stars are swept across the emulsion too rapidly. In this case, only six to eight very bright stars are photographically recorded. These star images are used to compute the camera orientation versus time. At least 20 stars are desired for adequate redundancy in the least squares computation of camera orientation. If the drift rate is too slow the images will tend to overlap or become too large. If the exposure time is extended, the images become elongated and too large, making comparator measurement difficult. When the exposure interval is too short, the images tend to overlap. Long exposure intervals result in less data and reduces the accuracy of the least squares computation. Poor balancing of the system causes slurring of the images. Thus, a precise combination of all factors is necessary for best results.

Some mention of what appears to be normal gyro-camera images is appropriate. Because of imperfect roll and pitch compensation, residual motions of the camera occur, giving the interrupted stellar traces a sine wave appearance. The amplitude and frequency of these sine wave traces depend upon the nature of the roll and pitch, the quality of stable platform compensation and the degree to which the system is balanced. The wave length is primarily dependent on the induced drift rate. With ideal roll and pitch compensation the interrupted star traces would appear as nearly straight lines. Quality data, however, can be extracted from the sinusoidal traces.

B Phase II - Simulator Tests

In Phase II tests, the gyro-stabilized camera was used to determine geodetic position. During these tests the gyro-stabilized camera was installed on the simulator which was mounted over a geodetic triangulation marker at Cape Kennedy. The gyro-stabilized camera along with five fixed stellar-oriented cameras observed a strobe light mounted on a C-131 aircraft flying over Cape Kennedy at an altitude of 18,000 ft. The test configuration is shown in Fig. 1.

Two separate tests were run on 31 August 1962. However, only one (Test No. 5595-2) provided usable data. The operating conditions and raw data obtained were as follows:

1. **Aircraft**

   a) Altitude = 18,000 ft
   b) Air Speed = 180 kts
   c) Course = SW to NE
   d) Strobe flash rate = 1 Pulse per 2 seconds

2. **Gyro-Stabilized Camera**

   a) Look Angle - Azimuth = 199°
   b) Look Angle - Elevation = 69°
   c) Focal Length = 210 mm
   d) F Stop = 4.2
   e) Drift rate = 0.028° per second
   f) Drift direction = Pitch (N to S)
   g) Exposure time = 0.4 second
   h) Exposure interval = 3.0 seconds

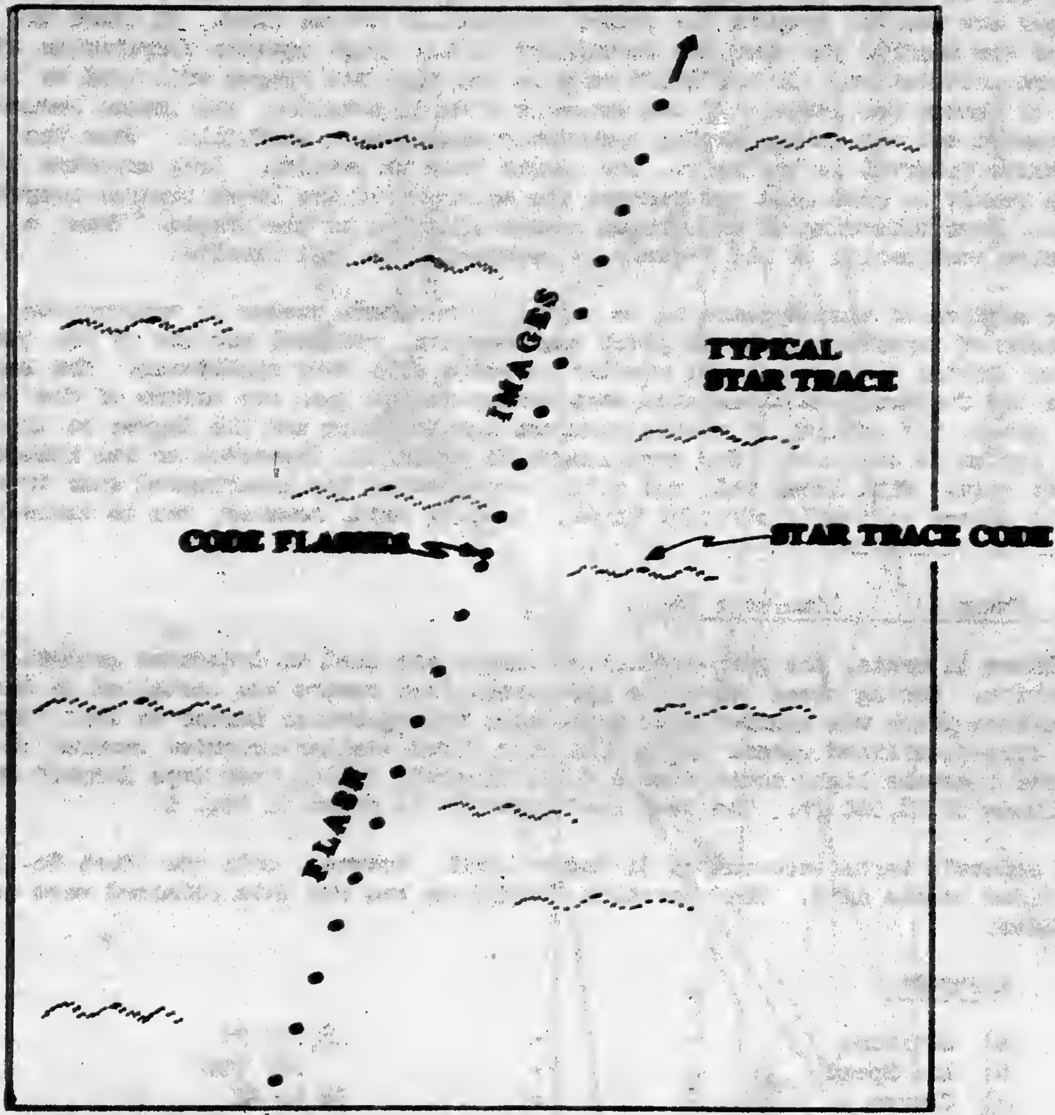

Fig. 2
SIMULATOR TEST SKETCH OF GYROCAMERA PHOTO-PLATE

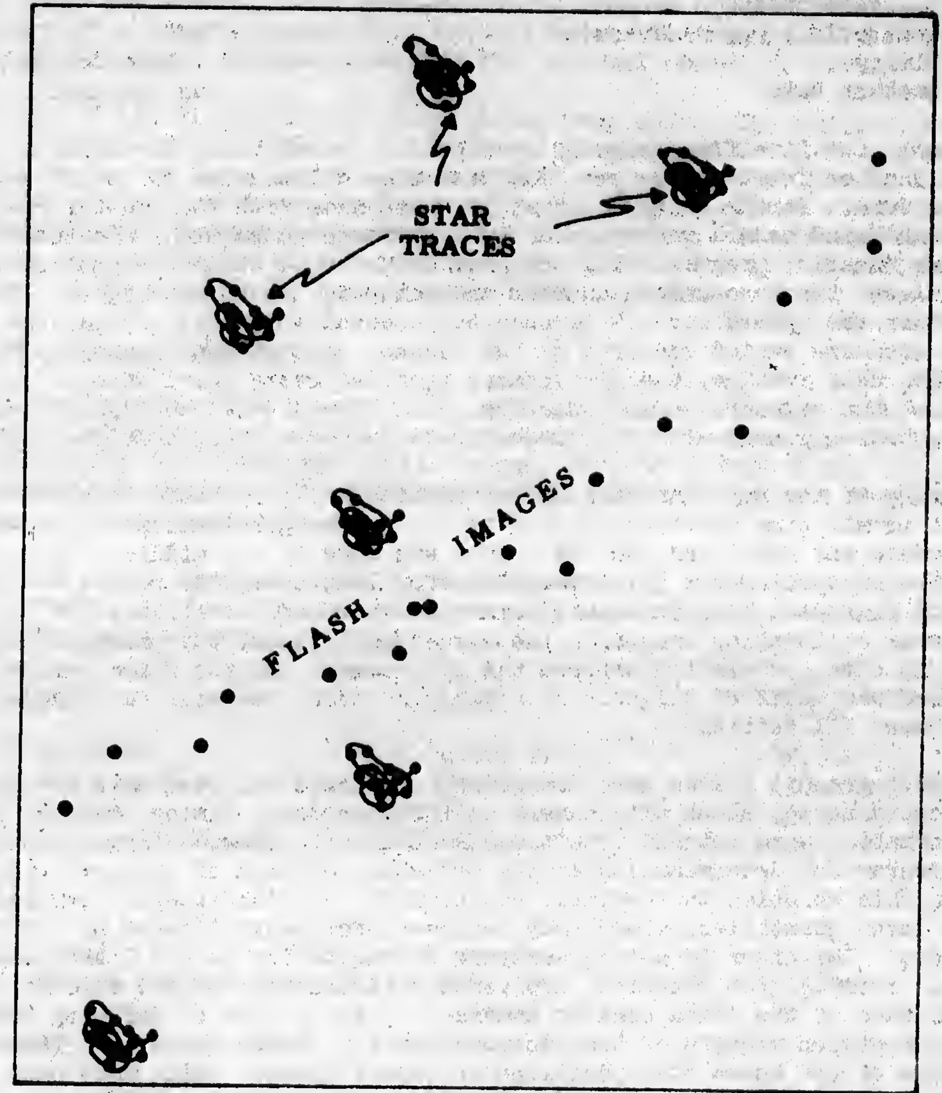

Fig. 3

SKETCH OF GYROCAMERA PHOTOGRAPHIC PLATE WITH POOR STABILIZATION AND NO INDUCED DRIFT

3. Recorded Data

a) Star traces (gyro-camera)	25
b) Discreet images per star trace (gyro-camera)	20
c) Data points (gyro-camera)	20
d) Timing	1 Msec accurecy
e) Weather data	as required

For this test, the five fixed cameras operated in normal mode, involving a preprogrammed shutter function for pre- and post-star calibration to obtain camera orientation data. Strobe flash observations were made with the shutter open. The gyro-stabilized camera was operated in a different mode with an automatic time-shutter function program. This program generates a control signal which opens and closes the gyro-camera shutter and actuates the strobe light. The camera shutter was opened for 0.4 seconds at 3-second intervals so that the middle of the exposure period occurred on the second. The strobe flashed on the second. With this routine, the gyro-camera recorded every third strobe flash at the middle of the exposure period (see Fig. 2). Fig. 3 shows how the photographic plate would have appeared with poor stabilization and no induced drift.

Special coding of the gyro-camera shutter function was programmed into the signal sequence to permit time correlation of data within the gyro-camera and between the gyro-camera and the fixed cameras. This was done at the middle of the observation period by opening the gyro-camera shutter long enough to record two consecutive strobe flashes. This procedure permitted excellent correlation of the flash and star calibration images on the gyro-camera plate, but presented some difficulty in time correlation between the gyro-camera and the fixed cameras. The procedure was modified slightly in subsequent tests and will be discussed under the Phase III testing.

After the photographic plates were developed, previewed and read on the comparator, the plate coordinates, along with camera calibration data, timing, weather and survey information were entered into the RCA "Ballistic Camera - Space Resection" computer program for determination of the geodetic position of the gyro-stabilized camera. In this solution the geodetic position of the fixed stellar cameras were entered as known quantities, since they had been previously located by accurate field surveys. The error in photogrammetric triangulation to the flash images were considered. Briefly, the solution involves a determination of the orientation of the optical axes of the fixed stellar cameras. This is done by defining the direction of the position vectors of the recorded stars in image space with respect to the direction of the known star positions in object space. Using this data and the image coordinates of the strobe flashes from all five fixed cameras, each flash point can be triangulated to obtain its spatial position. With the spatial position, the coordinates of the gyro-camera flash image and the orientation of the gyro-camera known, the geodetic position of the gyro-camera can be determined for each flash image recorded.

By virtue of the enduced drift, the gyro-camera orientation is continually changing as a function of time, whereas the fixed cameras have but one orientation. Accordingly, each group of stellar images recorded by the gyro-camera during an exposure period is used to determine a specific camera orientation that applies to the flash image formed during the same exposure period. This explains the reason for synchronizing the strobe flash to the center of the exposure period.

There is one main pitfall with this technique which so far has not appeared detrimental. That is, one cannot be absolutely sure that the computed gyro-camera orientation is completely valid at the instant of the strobe flash. If, in fact, the strobe does flash at exactly the middle of the expsoure period, and if the corresponding stellar images are symmetrical with respect to time (uniform drift rate) then the technique is valid. If this is not the case, some extrapolation is required to satisfy the simultaneity condition for the stellar and flash images. This problem can be avoided by such technique as observation of a passive satellite. In this case, simultaneity is achieved by exposing the stars and the satellite for an equal period of time.

An analysis of the simulator test results has been made. The results show the Circular Probable Error (CPE) of the computed position of the gyro-stabilized camera to be better than 1.22 ft. This result was much better than had been expected with the equipment used. The encouraging results of the simulator test increased interest in testing the equipment aboard ship. Of primary concern in ship testing is the added effect of ship's yaw on geodetic position accuracy and the effect of wave action on photographic image recording.

C Phase III - Ship Testing

The test configuration for Phase III is shown in Fig. 4. During these tests the gyro-stabilized camera was mounted on a deck near the stern of the AMR instrumentation ship Twin Falls Victory. This camera, along with five fixed stellar cameras, observed a strobe light mounted on a B-57 aircraft flying a prescribed pattern at 40,000 ft altitude.

Six separate tests were run on 22 April 1963, however, only two (Tests 2650-4,5) yielded enough data to provide a solution. The operating conditions and raw data obtained were as follows:

		Run 4	Run 5
1.	Aircraft		
	a) Altitude	40,000 ft	40,000 ft
	b) Air speed	400 kts	400 kts
	c) Course	0°	225°
	d) Strobe Flash Rate	one PPS	one PPS
2.	Ship		
	a) Speed	4 kts	4 kts
	b) Course (approx.)	168°	168°
3.	Weather		
	a) Sea state	1 to 2	1 to 2
	b) Wind direction	180°	180°
	c) Wind velocity	12 kts	12 kts
	d) Visibility	clear	clear
4.	Land-Based Cameras		
	a) Quantity (stellar)	5	5
	b) Focal length (stellar)	2 - 210 mm 3 - 300 mm	2 - 210 mm 3 - 300 mm
	c) Quantity (cinetheodolites)	4	4

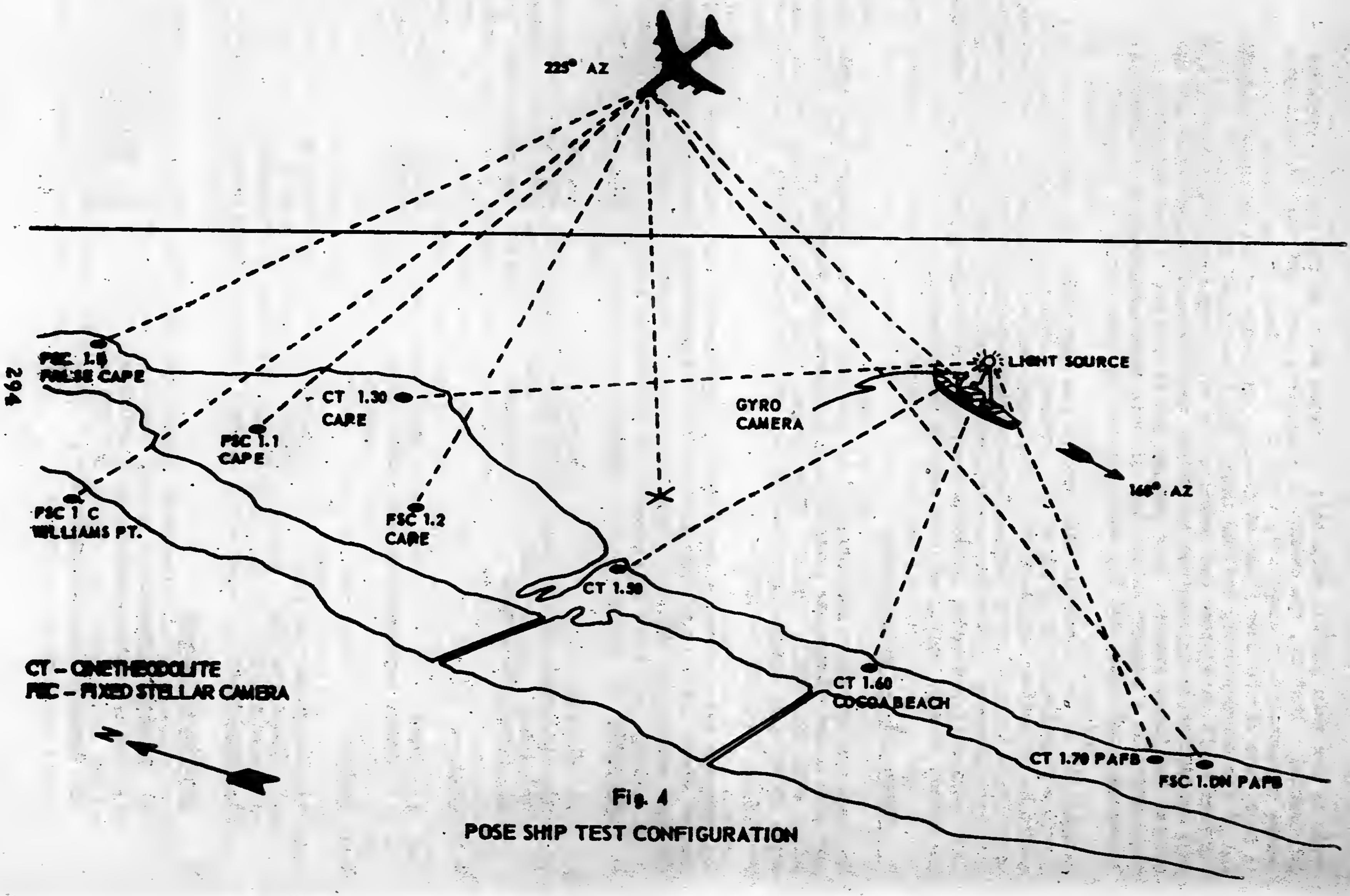

Fig. 4
POSE SHIP TEST CONFIGURATION

5. **Gyro-Stabilized Camera**	**Run 4**	**Run 5**
a) Look angle - Azimuth	316°	319°
b) Look angle - Elevation	63°	60°
c) Focal Length	210 mm	210 mm
d) F stop	4.2	4.2
e) Drift Rate	0.028° per second	0.028° per second
f) Drift Direction	Roll (E-W)	Pitch (N-S)
g) Time of observation (EST)	2302.5	2308
h) Exposure Time	0.4 second	0.4 second
i) Exposure Interval	4.0 seconds	4.0 seconds

6. **Recorded Data**

a) Star traces (gyro-camera)	17	14
b) Discreet images per star trace (gyro-camera)	29	29
c) Data Points (gyro-camera)	17	16
d) Fixed cameras (stellar, 3 - 300 mm, 2- 210 mm)	Good star and flash images	
e) Cinetheodolites	2 with dial readings and photo images. 2 with dial readings only	
f) Timing	1 Msec accuracy	
g) Weather	as required	
h) Ship Lorac Record	poor	poor
i) Ship attitude record	N/A	N/A
j) Ship speed record	N/A	N/A

During these tests, the Twin Falls Victory sailed on a course determined by Lorac navigation or by radar track from Cape Kennedy. At the same time, the B-57 aircraft was vectored over a given course by Cape radar. A knowledge of the aircraft position as well as the ship's position and heading versus time was essential to assure proper look angle pointing of the gyro-camera. A continuous light source, placed on the mast of the ship, served as the target for cinetheodolites' observations. The purpose of the cinetheodolite observations was to locate the ship accurately and continuously during a stellar camera observation period. The ship position thus obtained served as the standard against which the computed gyro-camera position of the ship was to be compared.

Before the test, the time for starting gyro-camera observations was based on ship's position which facilitated the setting of look angles into the gyro-camera. In practice, however, this was unsatisfactory. The aircraft could not be maneuvered into position at the proper time to assure that the strobe flashes would occur at the correct place in the gyro-camera field of view.

Therefore, the time of gyro-camera observations was pre-set on the basis of the anticipated position of the aircraft. Gyro-camera look angles were then based on the expected position and heading of the ship at the intended time of observation.

A modified method of coding was used to obtain star and flash image time correlation of the gyro-camera data and between the gyro- and fixed stellar cameras. With this sequence a series of discreet star images of nearly equal size for each star trace and evenly spaced flash images are formed on the camera plate. At the middle of the sequence, one gyro-camera exposure is increased to 2.4 seconds. During this 2.4 second exposure, three strobe flashes would normally occur at the 1 pps flash rate. However, the middle flash of this 3-flash group was deleted. Thus, one elongate star image is formed and two closely spaced flash images are recorded. The deleted flash which is notably absent on both the gyro-camera and the fixed stellar camera plates, permits time correlation between cameras. The elongate star image permits time correlation of the star and flash images on the gyro-camera plate. The gyro-camera photographic plates for this test are similar to that shown in Fig. 2.

The ship test was conducted under good weather conditions. This was both an advantage and a disadvantage. The relatively calm sea state reduced the ship roll, pitch and yaw motions to a low level. The gyro-camera could compensate for these motions rather easily, and resulted in the recording of good reduceable data. More pronounced wave action would have provided a better test of the system, especially the effect of yaw which cannot be compensated for by the stable platform.

The stellar camera data from these tests were also reduced by the "Space Resection" computer program to obtain ship's position. The cinetheodolite data was reduced separately to obtain light source position on the ship at one second intervals, on the second. The plan was to correct for the difference in position of the gyro-camera and the cinetheodolite target by combining their physical separation aboard ship and the ship's attitude data. Since the ship's attitude data was not recorded, the error analysis could not be done on a position basis.

A good analysis can be effected by comparing the computed separation of the gyro-camera and cinetheodolite target with the same separation, as determined by accurate theodolite survey when the ship was in port. The separation, as shown in the theodolite survey, was 159.872 ft, and was considered the measured standard for comparison purposes. The average computed separation for the two tests was 161.760 ft. The computed separation was found by differencing the computed coordinates of the gyro-camera and the cinetheodolite target for each flash image recorded. The mean error of the separation was 1.888 ft. The combined circular probable error (CPE) of the two runs was 2.06 ft. This indicates that a ship can be located to a 2.06 ft accuracy (50 percent circular probability) under these test conditions. The result is considered quite good.

Consider roughly what this accuracy represents when observing a satellite of known position at a slant range of 1000 nm. During the Phase III tests, the average distance of the aircraft strobe light from the gyro-camera was 48,800 feet and the CPE of positioning the gyro-camera was 2.06 feet. This corresponds to an angular error whose tangent is 2.06/48,800 feet or approximately 8.5 arcseconds. Satellite position can be determined to an accuracy of plus/minus 50 feet or better by simultaneous observations with stellar oriented cameras. Taking the uncertainty of the satellite position, and the angular error of the gyro-camera and other factors into account, location of a ship to a CPE of 300 feet or better would be possible with the equipment and techniques used and under the weather conditions during the Phase III tests.

OPERATIONAL METHODS

Four operational targets can be used to determine ship's position with the gyro-camera. Target selection depends on the location of the ship to be positioned. If it is relatively close to land, aricraft targets may suffice. If far at sea, missile, active or passive satellite targets can be used. Orbital observation with a single gyro-camera represents the simplest and most direct method of determining ship's position, however, the orbital position of the target must be well defined. The simultaneous method of observation is presently the most accurate and reliable for AMR needs.

Some mention should be made concerning ships position and how it can be transferred to hydrophones and transponders for a permanent ocean bottom reference. Since the ship is usually in motion, a different position will be determined for each target image recorded. All these positions must be used in a least squares solution which will be better than any one of the individual positions determined. This can be done by correlating ship's position as established by the gyro-camera, with the ship's position relative to the transponders as determined by transponder interrogation. By equating the change rate of ships position, found by the gyro-camera, to the rate of change obtained from transponder interrogation, a set of equations can be derived for least squares determination of transponder position. Observational data from several passes of the target will permit better statistical adjustment, thus more accurate transponder positioning.

SOLAR ENERGY CONVERSION BY THERMAL DESTRUCTION OF SUPERCONDUCTIVITY

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Summary

A method is proposed for converting solar radiation directly to electrical energy in space by utilizing the change of state of a superconductor. Apparatus is described whereby intermittent exposure to a concentrated beam of solar radiation causes a magnetic shield to fluctuate between the superconductive state (zero permeability) and the normal state (unity permeability) with resultant flux changes converted to electrical energy by conventional induction methods.

A thermal embodiment of Lenz's law, akin to the back torque of a conventional generator or the counter current in the primary of a transformer, is postulated. Such a reaction is demanded by energy conservation principles, although in the present case it can be exerted only against an input of purely thermal energy.

In a heat engine working between heat quantities represented by intercepted solar flux at the upper limit and the black cold of space at the lower limit, a theoretical thermal efficiency in excess of 99% is possible. In order to realize this phenomenal efficiency in a heat engine with a radiator of practical size, a closed compound cycle is proposed in which a portion of the power output is used to drive a refrigerator. By compounding the cycle the net thermal efficiency is reduced but the effectiveness of the system remains unchanged, provided only that a part of the output of the converter be utilized in the cooling of other cryogenic devices such as superconductive delay lines, lasers, IR sensors etc., the balance being available for general purposes. The heat extracted from all cryogenic equipment, including the converter itself, can then be radiated away at a temperature high enough to benefit from the fourth power law.

A practical form of the converter is described, with particular reference to the problem of eddy-current losses, which are unusually severe at very low temperatures.

Introduction

Superconductivity has many potential applications in space technology. Long regarded as remote and forbidden territory, the superconductive state is now being investigated routinely in many laboratories. From these investigations have come some remarkable devices: the cryotron, a tiny ultra-fast computer element; the superconductive electromagnet, capable of producing fields of hundreds of thousands of gauss in a magnet weighing pounds instead of tons; the superconductive delay line which operates without loss, and the frictionless gyro bearing spinning on a reflected magnetic field. In addition, many components such as IR sensors, lasers and masers benefit from the low temperatures associated with superconductivity.

In space, where intense heat and intense cold co-exist, we could now make good use of Maxwell's Demon in separating galactic radiation from solar radiation by degree of attenuation. Lacking such fanciful means we must resort to brute force engineering to effect a separation. The advantages of sustained cryogenic temperatures in space are nevertheless so compelling that efforts are presently being made to develop a flyable helium liquifier. Machinery of this kind, however, imposes an inordinate demand upon the already hard-pressed electrical generating systems now in use or in immediate prospect. A refrigerator is basically a heat engine run backward. If it is powered by another heat engine operating in the conventional 30% area, very little backfield is left. This is especially true if the generator is also expected to furnish electrical power for general purposes. But if it were possible to endow the primary system with a thermal efficiency in the neighborhood of 90%, the refrigerator could run back up the scale, pay dividends in the form of cooling, and leave the primary system at a point where conventional conversion methods expect to begin. It is the purpose of this presentation to outline one method by which an answer to this problem in cryogenics may be found within the realm of cryogenics itself.

Magnetic Properties of a Superconductor

A superconductor posesses unique magneto-thermal properties which may be applied to the purposes of energy conversion. In fact, heat and magnetism are so closely related in the change of state of a superconductor that their interaction has been the subject of extensive treatment in the literature. For example, it has been found that a latent heat of transition accompanies the change of state of a superconductor in the presence of an external magnetic field; latent heat effects are totally lacking when the transition occurs in the absence of an applied field. Since magnetism is directly related to electricity, it follows that a triple play is theoretically possible: heat to magnetism to electricity, all within a solid state medium. It is true that classical experiments in superconductivity performed with free magnetic fields involve only minute quantities of energy. But we are not going to concern ourselves with magnetic fields linked to nothing more demanding than the quiet air of a laboratory. All that is needed to produce a highly reactive magnetic field is simply to put it to work. The energy involved in the manipulation of a working magnetic field is worthy of attention from the standpoint of energy conversion even though its rowdy behavior may have caused it to be banished from the laboratory.

The property of a superconductor which appears to lend itself most readily to the purpose of energy conversion is that of zero magnetic permeability. When a magnetic field is applied to a superconductor an electric current is induced at the surface of the metal. The induced current encounters no resistance and hence continues to flow even after a state of magnetic equilibrium is reached. Moreover, the magnetic field accompanying the induced current opposes the external field in both intensity and polarity. The applied field therefore cannot penetrate the metal, except to a depth of the order of millionths of a centimeter.

This phenomenon is illustrated in a striking way by the famous floating magnet experiment, as shown in Figure 1. If a bar magnet is placed above a superconductor, the persistent currents generated by its approach give rise to an opposing magnetic field which precisely matches that of the magnet. The magnet will then float forever upon its own reflected field. The lines of force cannot penetrate the superconductor, yet each must form a closed loop. Therefore, these magnetic lines of force will be parallel to the surface of the superconductor and must also be somewhat compressed because they occupy a smaller cross-sectional area than they would normally occupy in free space. This compression of the lines of force, or in other words, intensity of the incident field, cannot exceed a certain critical value without causing breakdown of the superconductive state. For each of the superconductive metals at a given temperature there exists a critical magnetic field.

Figure 2 shows the relationship between temperature and critical field (H_c) for niobium. The metal niobium was chosen for illustration because it will maintain the superconductive state in the presence of relatively intense fields. It will be seen, for example, that at a temperature of 5 degrees K, corresponding roughly to the boiling point of helium, niobium will resist penetration by a magnetic field of 1800 gauss. Certain alloys such as niobium-zirconium will remain superconductive in the presence of much higher fields but do not exhibit zero permeability over their entire surfaces due to the inclusion of normal areas. As conductors of supercurrents such alloys are useful since the superconductive portions are of filamentary form and therefore maintain continuity. To serve as a magnetic insulator, however, a metal must be of the highest purity.

Suppose now that we bend the bar magnet of Figure 1 into the form of a "C" with the superconductive sheet interposed in the gap as shown in Figure 3. This exercise in topology has caused no change in the magnetic circuit except for a variation in the leakage pattern. The flux lines are still repelled by the superconductive surface.[1] The magnet is now a folded bar magnet but the reluctance of the circuit remains practically the same. Each pole is reflected from an opposite side of the superconductor but the lines of force are still parallel to the surface and they must travel almost as far to get home.

If the superconductive barrier were to be removed, we would then have a true "C" magnet with the greatly reduced reluctance of that configuration. This can be done in effect simply by heating a central area of the barrier until it returns to the normal state. The source of heat could well be a concentrated beam of solar radiation. Or it could be a laser beam carrying power from one of a series of generating stations on earth. In any case, restoration of the normal gap length will reduce the reluctance of the magnetic circuit by a very considerable factor and increase the flux density within the magnet by the same factor. If a coil is linked with this change in flux as shown in Figure 4, an emf will be induced in the coil and a useful electric current will be generated. Moreover, Lenz's law insures that the input energy will be required to match the output energy as it does in every other member of the family of electro-magnetic generators.

In order to function as a useful energy conversion means, the process must be repeatable. A series of pulses or cycles must be generated, preferably at a fairly high frequency. In theory this does not entail any difficulty since the superconductive phase transition is thermodynamically reversible. If the metal is cooled through a very small threshold temperature range it will return to the superconductive state. Of course, no energy can be converted during this phase change because none is received. Any attempt to impose a load upon the transition from normal to superconductive would undoubtedly block the transition. Such an occurrence has in fact been reported by von Laue, although the effect was accidental and very little significance was attached to it.[2] In Figure 4 a diode is shown in series with the output coil to prevent the development of a back mmf during the reverse transition. Lenz's law can work both for and against us; if it can demand energy when energy is available and eager to be converted, it can oppose a flux change equally as vigorously when nothing more is forthcoming than the exceedingly feeble free energy difference between the two states of the medium itself. It should be noted, however, that no power on earth or in space can stop the transition from superconductive to normal. If this were not the case, the superconductive state could be carried up to room temperature simply by threatening it with a job.

Even a free magnetic field relieved of the burden of an inductive load can interfere with the transition from normal to superconductive to a certain extent. In 1933, Meissner and Ochsenfeld established the fact that a superconductor will not only resist penetration by a magnetic field but will actually expel a field that is present during the transition, provided that the field strength is less than critical.[3] But the expulsion of an existing field is subject to imperfections in the form of flux entrapment, the configuration shown in Figure 3 being particularly susceptible. To facilitate the return of the metal to the superconductive state the field in the gap should be reduced as much as possible immediately prior to the transition. Since the output coil has in effect been open-circuited by the blocking effect of the diode, the field in the gap can be reduced or even eliminated during this phase of the cycle by a suitable shunting means at no greater cost than that of hysteresis and eddy-current losses. This kind of trade-off is not uncommon in energy conversion devices. The four-cycle gasoline engine, for example, goes through a very elaborate and energy consuming mechanical routine before returning to its single power stroke.

Design Concepts

It is apparent that the rudimentary form of the device as shown in Figure 4 does not lend itself readily to cyclic operation. Even with the output coil open-circuited it would be difficult to remove the field in the gap of a permanent magnet without having to contend with some rather violent reactions. These reactions could be nulled out by complicated energy-recovery machinery, but it is always advisable to avoid machinery, especially in the space environment. However, before proceding to the description of a more sophisticated design, it would be well to isolate and analyze what may be called the power stroke of the cycle.

To do this it will be necessary to assume certain dimensions. Let the pole diameter be 1" and the normal gap length 2" as shown at G_1 in Figure 4. Using niobium as a superconductor at a temperature of 5 degrees K, the field strength in the gap must not exceed 1800 gauss. If the magnet is of Alnico V and is properly designed to operate at a point on the demagnetization curve at which the energy product is a maximum, the flux density within the magnet will be 10,500 gauss when the barrier is in the normal state. With the barrier in the superconductive state there will be no gap at all, properly speaking, but merely a complex flux leakage pattern. For purposes of calculation the leakage path can be represented by a hypothetical gap G_2 having a length of 6 1/2" based upon a barrier diameter of 6". The relationship between gap length and reluctance is highly indeterminate in any magnetic circuit. Normally about 90% of the reluctance of the circuit resides in the gap and hence any change in gap length has a profound and disproportionate effect upon the reluctance. To be conservative in this case, we will assume that the reluctance varies linearly with respect to gap length. The change in flux density varies directly with reluctance; therefore, a change in gap length from 6 1/2" to 2" under the effect of thermal destruction of a superconductive barrier will result in a flux change of 10,500 x 2/6.5 or 3240 gauss.

If this flux change is not put to work it represents very little energy; no more, in fact, than the energy represented by the original distortion of the free field. The distinction between a working flux change and one that merely exercises its little domains is forcibly illustrated by the flux changes occurring in the core of a power transformer. With the secondary open-circuited the core can be saturated by an exciting current of only about 3% of the rated load current. With an output current flowing in the secondary, the same flux change will transport an enormously greater amount of energy from the source. This device is in effect a transformer with an exotic heat-actuated primary and a conventional secondary.

Now let us see how much energy can be transported by a working flux change of 3240 gauss. Assuming a uniform change in flux, the resultant emf can be calculated by Faraday's formula,

$$E = \frac{\Delta B\, A\, N}{t} \times 10^{-8}$$

in which ΔB is the flux change in gauss, A is the area of the core in square centimeters, N is the number of turns in the coil and t is the interval of time in which the flux change takes place. A coil consisting of 150 turns of #18 wire is arbitrarily assumed. The time required for the transition depends upon the incident heat energy, the strength of the applied field and the thermal mass of the medium. In this case, using a film 1/10000" thick, a pulse duration of .001 sec. does not appear to be unreasonable, considering the fact that in a cryotron the transition occurs in about 15 billionths of a second. Substituting these values, we have

$$E = \frac{3240\,(5.07)\,150}{.001 \times 10^{8}} = 24.6 \text{ Volts}$$

Since heat is normally absorbed at a uniform rate, there is reason to believe that the flux change will be uniform with time and that the output will therefore be pure d-c. In this respect the device is comparable to a homopolar generator.

Taking full advantage of the cryogenic facilities which this converter will presumably make available, the coil can be maintained at an average temperature of 127 degrees K by means of a liquid nitrogen jacket, assuming a temperature rise of 50 degrees. At that temperature its resistance will be about .09 ohm. The short-circuit output will then be

$$P = \frac{24.6^2}{.09} = 6.6 \text{ KW}$$

An output of this order of magnitude in a converter weighing only a few pounds indicates the possibility of an interesting future. Upon such a premise we may be justified in giving some attention to a more practical form of the device.

Figure 5 shows an arrangement having two gaps, each with its own barrier, 1 and 2, together with a branching system of ferrite pole pieces connected to a permanent magnet 3. The barriers 1 and 2 are alternately irradiated by means of a rotating mirror not shown in the sketch. Thus the reluctance of one branch increases as that of the other decreases, the flux changes being confined to the pole pieces which serve as cores for output coils 4 and 5. In this way the total flux in the magnetic circuit remains the same, although in each branch it may fluctuate through a range that could not be tolerated in even the most highly stabilized permanent magnet. The magnet 3 furnishes what may be called magnetic potential and is not affected by the flux changes in the pole pieces except perhaps to a very limited extent. The barriers 1 and 2 are maintained at superconductive temperature by contact with liquid helium. Control windings, shown at 6 and 7, are wound in such direction as to buck the magnetic flux in each branch.

We now have the makings of a cycle. Let us start with barrier 1 in the superconducting state at the beginning of the irradiated phase. Barrier 2 is in the normal state. Control windings 6 and 7 are open-circuited. Coils 4 and 5 are connected to the load. As barrier 1 passes to the normal state the reluctance of the associated branch is reduced and a current is induced in coils 4 as a result of the expanding field. The flux density in the opposite branch is reduced as the flux tends to equalize and a current is then induced in coils 5 due to the collapsing field in that branch. Coils 4 and 5 are of course wound in series aiding fashion. When barrier 1 has completed its transition to the normal state, or perhaps slightly before, all coils are open-circuited by a suitable switching means. During the time that the energy beam is moving from barrier 1 to barrier 2, control winding 7 is energized, removing the field from barrier 2 which then snaps into the superconducting state. This transition requires no energy whatever in the absence of a magnetic field and occurs in practically zero time. It should be noted that the current in control winding 7 is not working against any load other than the resistance of the coil and the reaction of the core material. It is comparable to the exciting current of an unloaded transformer and, considering its transient nature, probably represents an even smaller fraction of the total output. With the de-energizing of control winding 7 coils 4 and 5 are reconnected to the load through a solid-state commutator and the cycle is then repeated.

Solar energy is supplied in space at the rate of about 10^3 watts/m^2. Taking as a goal the 6.6 kilowatts of the power impulse of the prototype design which the improved form should provide almost continuously, a collector about ten feet in diameter will be required, allowing a collector efficiency of 75%. A concentrated beam 2" in diameter (to conform to the dimensions of the magnet gap) will then provide an energy flux of 270 watts/cm^2. This corresponds to the temperature, T_1, of an equivalent black body close to the superconducting surface, where:

$$T_1 = \sqrt[4]{\frac{270}{5.8 \times 10^{-12}}} = 2600^\circ K$$

Waste heat is rejected to the liquid helium coolant at a temperature, T_2, of about 5 degrees K. The thermodynamic efficiency of the converter is therefore

$$\eta = \frac{T_1 - T_2}{T_1} = 997$$

Thus, only 0.3 of 1% of the input heat is unavailable for conversion and must be rejected to the coolant. In terms of electrical power this represents a loss of only 3 watts per kilowatt.

At first glance it seems highly improbable that a film of niobium .0001" thick mounted on a thin substrate and subjected to a temperature of 2600 degrees at one face and 5 degrees at the other would survive for even a small fraction of a second. Ordinarily it wouldn't. In a passive magnetic field the consequences would be catastrophic since the latent heat of transition in the presence of such a field is negligible. In the case of a working magnetic field there is a vital difference. The input of heat is reflected directly and immediately in an output of electrical energy. Over 99% of the incident heat energy is available for this purpose. Obviously, the same unit-of-radiant energy that contributes to a changing magnetic flux and is withdrawn by induction to do honest work elsewhere cannot at the same time remain behind for the purpose of heating up a metal film.

Ideally, an energy conversion medium serves only as a stepping-stone. In most heat engines the conversion process leaves behind it a trail of violence, stepping-stones that are melted down and recast for every erg that goes from one job to another. Yet there is no basic immutable reason for the working medium even to change temperature, except as required to dispose of unavailable heat. The intimate relationship between heat and magnetism in the phase change of a superconductor is such that an energy exchange can occur without any significant macroscopic change in the medium. Such an effect is rendered credible by the fact that the superconductive state of a metal does not differ physically from the normal state in any observable respect. The conversion of heat energy to electrical energy by means of the closely related magneto-thermal properties of a superconductor therefore does not require the molecular gyrations and consequent involvement of mass and volume that characterizes the alien working fluid of a conventional turbo-generator. This isolation and direct combination of properties contributing to thermo-electric conversion and only to thermo-electric conversion holds forth the promise of a compactness and power/weight ratio impossible even to approach by conventional means.

In any electromagnetic generator a reaction must exist which requires the input energy to match the output, except for the usual losses. The law of conservation of energy demands such a reaction. The back torque exerted by the shaft of a mechanically driven generator imposes a load upon the prime mover precisely proportional to the output of the generator. Similarly, the back emf in the secondary of a transformer induces a counter-current in the primary which opposes the input to the precise degree necessary to insure that conservation laws are satisfied. This requirement is encapsulated by Lenz's law, which states that any electric current initiated by a change in magnetic flux must react in such manner as to oppose the cause of the flux change. But in the present case the cause of the flux change is an input of thermal energy. The mechanism by which a back emf can react against a purely thermal input of radiant energy and require it to deliver full measure is at present unknown, at least to the writer. Certainly it could not be more complex than the series of energy exchanges by which the interaction between a coil and a magnetic field in a conventional steam driven generator is ultimately manifested in a cooling effect upon some distant flames which in turn gain their energy from a sun that set a million years ago. In the present case, a detailed analysis of the processes involved in the change of state of a superconductor under the effect of a reactive magnetic field is beyond the scope of this paper. Many of the writer's consultants have shown some interest in the theoretical aspects of the problem; eventually some light may be thrown upon the manner in which Lenz's law is manifested in this generator.

Ironically, one of the most obstructive roadblocks in the course of development of this idea arose from the high thermal efficiency of the device. Even though the amount of heat that must be rejected is very small the temperature difference between the coolant and the sink is also very small, which makes it difficult to get rid of the heat. In space it is much easier to radiate away kilowatts in the form of waste heat at the temperature of liquid mercury than it is to dispose of a fraction of a watt at the temperature of liquid helium. It is a case of Carnot vs. Stefan-Boltzmann. We can of course give the decision to Stefan-Boltzmann and commit ourselves for all time to the use of a high-temperature medium at a maximum thermal efficiency of about 35%. Or we can take the bolder course and try to invent our way around the problem.

If the temperature at which waste heat is rejected can be increased, the rate of radiation will increase as the ratio of the difference between the fourth powers of the temperatures of radiator and sink. This is in accordance with the Stefan-Boltzmann radiation law. On the other hand the expression for Carnot efficiency takes no heed of the use to which the work is put. Thus a heat engine can be caused to drive a refrigerator without losing a single percentage point of its original efficiency. The net efficiency of the compound system will be reduced, however, because the temperature, T_2, at which heat is ultimately rejected will be increased. But if one happens to need the services of a refrigerator (helium liquifier in this case) and also wishes to take advantage of the fourth power law, the combination is a particularly fortunate one. Moreover, when the primary system operates at a thermal efficiency in excess of 99% a considerable degree of freedom in the allocation of the output is permitted. For example, a cooler can be designed to draw 40% of the output of the converter, leaving 60% to be distributed as electrical power. If a smaller radiator is desired, together with the extraction of more heat from cryogenic devices, a proportion of 70% for cooling and 30% for electrical power may be chosen. Of course the efficiency of the cooler must be taken into account, but the overall loss is no greater when the cooler is compounded with the generator than when each goes its separate way.

With power in the kilowatt range available for closed-cycle cryogenic cooling, many superconductive devices now barely dreamed of will be possible. For example, an entire spacecraft can be shielded against dangerous radiation by sheathing it with an intense magnetic field, a field of the order of hundreds of thousands of gauss. Such an expedient would be impossible without the benefit of enormous currents running losslessly and eternally in threadlike wires. The space technology of the future will most certainly rely heavily upon supercurrents; devices unknown today will be conceived to meet problems of which we are not even aware. The ability to maintain superconductivity in space on a large scale will in itself give rise to many advances. The unique properties of the superfluid state may also find exploitation when the means for maintaining this state of matter is readily at hand.

Any approach to the objective of energy conversion in space must take into account the effects of the environment. The method proposed herein is no exception; at very low temperatures, materials behave quite differently than at room temperatures. For example, at cryogenic temperatures eddy current losses are very severe due to increased electrical conductivity of the core. The use of a ferrite material will help to alleviate this condition at the expense of increased bulk but with no increase in net weight. Further improvement may be possible by using a laminated ferrite. The core could be pressed in the green state with the ferrite contained between very thin strips of insulating material and the entire assembly then sintered. This improvement, if such it is, represents only one avenue of attack. There could be many.

Concluding Remarks

At the present time this method of energy conversion exists only in the mind of its inventor and in the files of the Patent Office. As a system it has not been physically developed beyond the performance of some simple experiments using superconductive lead and undertaken only for the purpose of demonstrating the process from a scientific standpoint. The experiments were successful and are serving as a basis for further investigation of a highly theoretical nature. In a practical way nothing has yet been done to provide experimental evidence of the feasibility of the system as a whole. Lacking such evidence, the system concept of the idea must necessarily hang upon a rather tenuous thread of logic. At this precarious point in its history, however, it does have one benefit to offer. Almost every energy conversion method now under development has filtered through many minds and in most cases many generations of minds. Improvement becomes not only increasingly difficult but less rewarding. In the present case a totally new method of converting heat to electricity is offered. Every major improvement has yet to be conceived. Despite its lack of ancestry and the fact that the sum total of its literature is represented by this one paper, the proposed method may open a promising field of investigation to the few who are not afraid of being alone in the dark.

Acknowledgments

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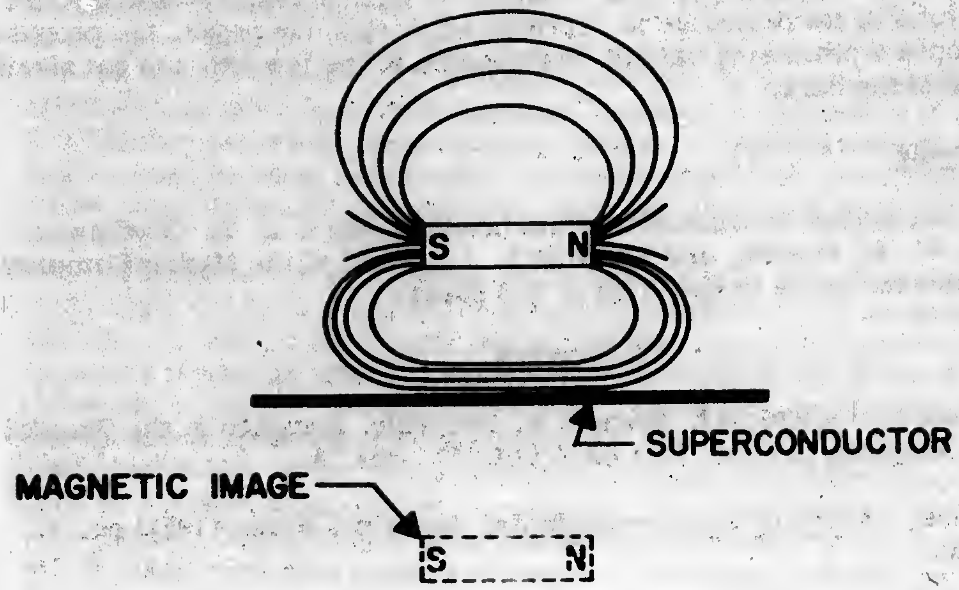

FIGURE 1

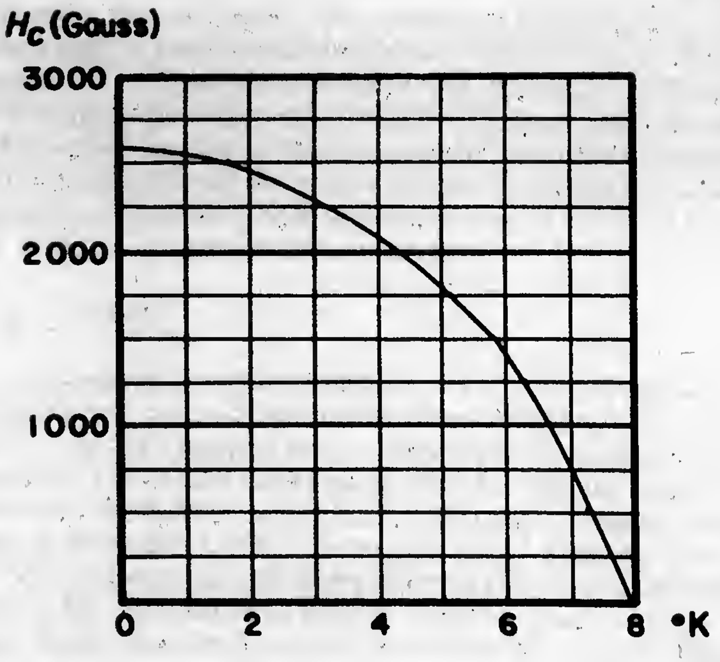

FIGURE 2

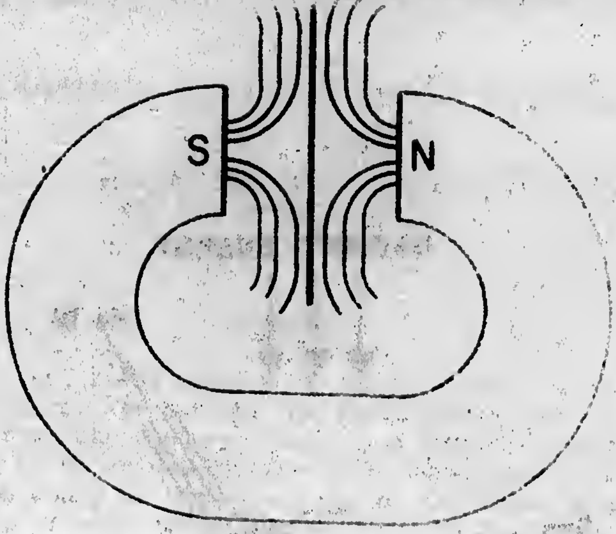

FIGURE 3

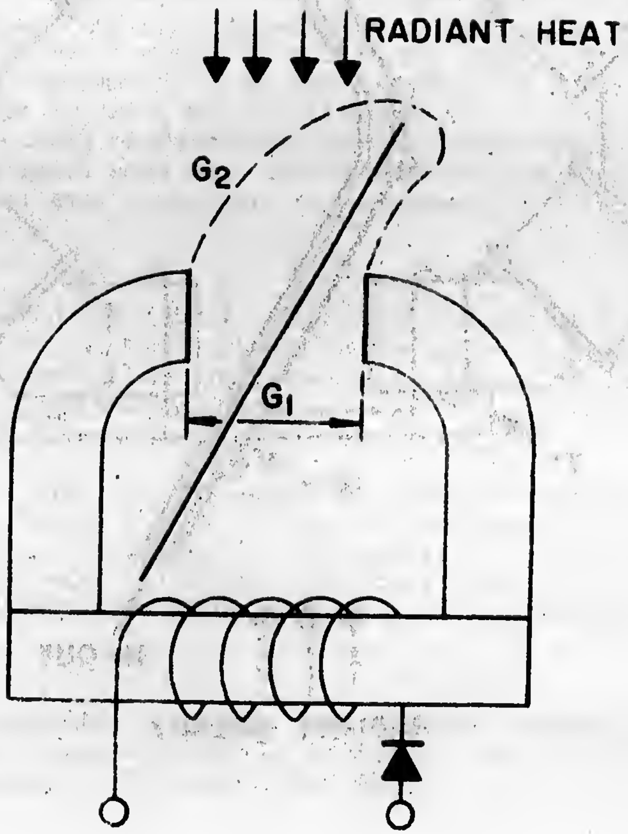

FIGURE 4

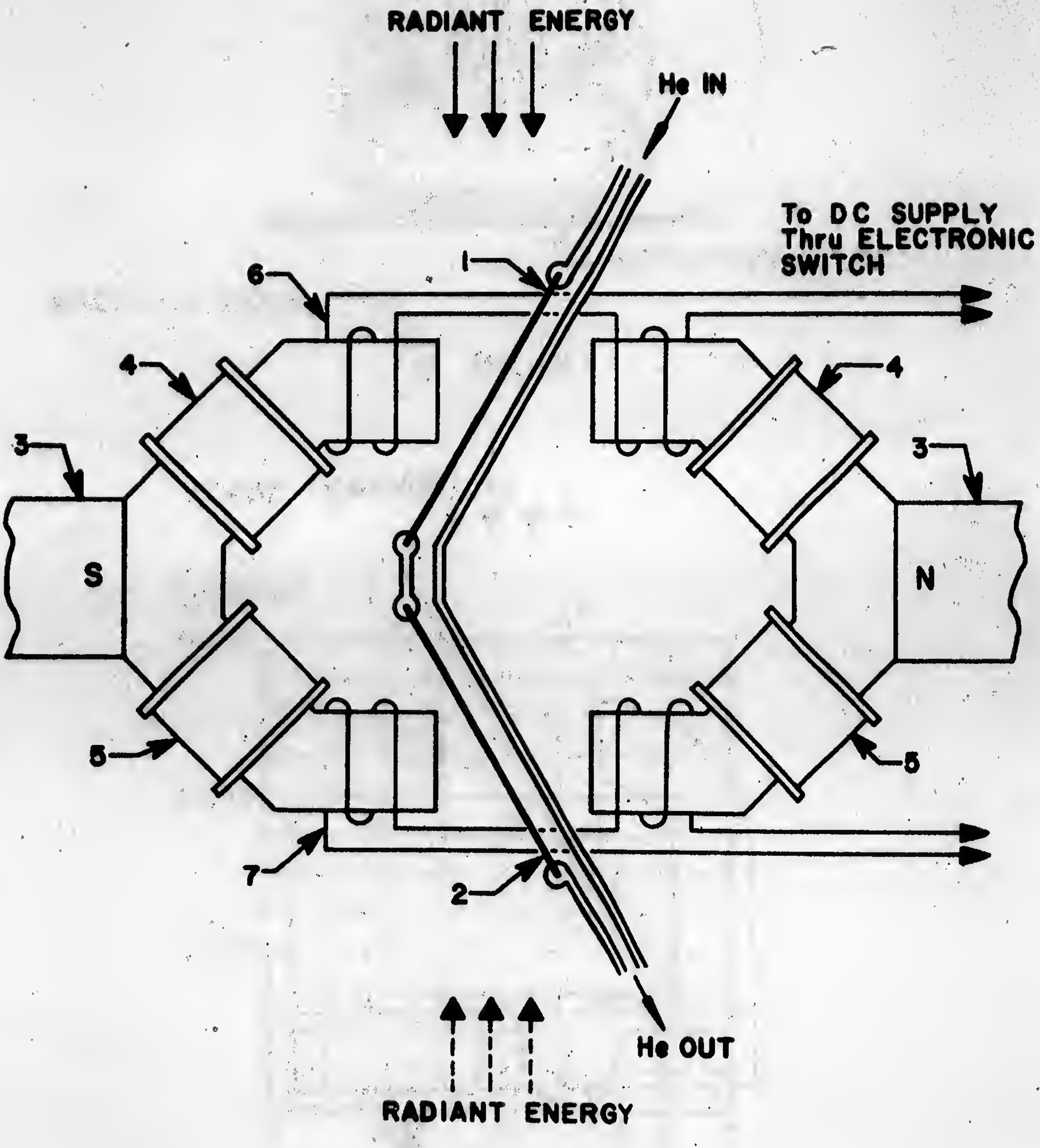
RADIANT ENERGY
He IN
To DC SUPPLY
Thru ELECTRONIC
SWITCH
1
6
4
4
3
3
S
N
5
5
7
2
He OUT
RADIANT ENERGY

FIGURE 5

SOLAR GENERATION OF ELECTRIC POWER AT LUNAR POLE

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SUMMARY

A theoretical investigation of possibilities for an electric power station, appropriate for installation and use on the moon, late in the 20th century, leads to a tentative decision in favor of a solar driven Rankine cycle turboelectric system. Such a station would be located at one of the lunar poles, to take advantage of continuous sunlight for the collector, and continuous shade for the radiator, in close proximity. Gross thermodynamic characteristics of the system are investigated in a preliminary way.

INTRODUCTION

The present paper suggests one possible solution to the problem of continuously generating large quantities of electric power on the surface of the moon. It is not claimed to be an optimum solution, but is rather an outline of a system concept which would seem to merit serious consideration if certain basic assumptions turn out, in the light of future development, to be correct. The system concept, while perhaps novel when taken in its entirety, comprises nothing very new in the way of subsystems, assemblies, or components.

Ground Rules

Our study has to do with a power plant which might be built 20 or 30 years from now. Accordingly, our calculations are all in the nature of rough approximations. Thermodynamic computations are based on the usual first order idealizations --- pure adiabatic expansion of gases in a turbine, frictionless flow (zero pressure drop) in tubes and conduits, and all the rest.

No special allowance is made for the advances in power engineering technology which are bound to take place between now and the time when such a plant might be built. This paper aims to show up certain possibilities which will almost surely exist. If in the next few decades still more attractive possibilities are brought forward, so much the better; but that is not our present concern.

Basic Assumptions

1. There will be established a more or less permanent manned station, somewhere on the moon, before the end of the present century. There may even be several stations, in different locations.

2. Any such manned station will require a continuous supply of electric power, in the range of 100 to 1000 kw.

3. It will prove both feasible and advantageous to locate at least one such base in the near vicinity of the lunar north pole of south pole.

4. Natural resources locally available on or near the lunar surface will have been exploited, and a technology will have been developed, to a point such that simple metal fabrication and construction projects can be carried on with a minimum of support from earth. Only special equipment and materials (tools, instruments, optics, special alloys, etc.) will need to be brought from earth.

DISCUSSION OF ASSUMPTIONS

The above are perhaps bold assumptions; but the published literature affords abundant support, particularly for the first three. Many writers have argued the need for a "lunar colony". (See refs. 1-8, also bibliography in refs. 2 and 4, particularly). Especially forceful is the recent review by Arthur C. Clarke[5], citing the benefits to be expected in several widely differing categories:

vacuum technology
radio astronomy (lunar farside)
logistics and supply to earth orbiting vehicles
base of operations for further exploration of solar system

Among other interesting proposals by Clarke is that of a moon-based electrodynamic launching system[1, 5]. This idea has been further developed by Escher[9]. Instantaneous electric power requirement for such a system, however, is extremely high (up to 10^9 watts or higher, per earth-ton of gross mass, accelerated to escape speed at 50 gravities), and would obviously require some kind of short term energy storage scheme, as well as a generating plant having a capacity suitably matched to that of storage device. The papers by Clarke and by Escher, in suggesting the need for large amounts of electric power on the moon, have provided a positive stimulus for the present study. Our proposal, however, falls far short of meeting the requirement implied by their electrical launch device.

Salisbury and Campen[23] and, more recently, J. Green[8] have considered in detail the relative merits of different possible locations for a permanent lunar colony. Green and co-authors[7] have put forward the idea of a base at one of the lunar poles. They have, indeed, already pointed out a number of the factors which are important to the subject being treated in the present paper.

The general problem of designing a lunar power station can be approached in at least 2 different ways:

(a) One may first select the site for the lunar colony or base, thinking chiefly of the functions the base is to perform, then decide on an optimum scheme to supply power to a base in that particular location.

(b) One may determine what kind of electric power system, somewhere on the moon, would yield the greatest utility per dollar spent, then feed the result into the process of making a decision as to the site of the lunar base.

This paper takes the second approach. Our concern is with the situation which develops after the early phases of lunar exploration have been completed. The need will no longer be for a power station which is highly mobile, and neatly self contained. Rather, to achieve maximum utility per dollar, one should seek to maximize the steady output per unit weight of equipment and material to be boosted from earth.

The endeavor to maximize this particular quantity leads to an overall design which differs markedly from that of power plants in the more usual categories --- Earth-based public utility, manned or unmanned spacecraft. It may differ, too, from the designs appropriate to the very early permanent stations on the moon --- stations deriving their entire support from earth.

Our treatment, while neither rigorous nor exhaustive, does lead to a tentative conclusion in favor of a solar-powered rankine cycle turboelectric installation at one or the other of the lunar poles. The chain of reasoning is set forth in the next section.

STEPS TOWARD SYSTEM DEFINITION

1. We recognize at least four possible prime sources of energy:

    a. Chemical fuels
    b. Heat from Lunar interior
    c. Solar energy
    d. Nuclear fuels

    Both (a) and (b) seem unattractive. Present knowledge is not sufficient to warrant outright rejection of either; but the present expectation is strongly negative. Suitable unreacted chemicals probably do not exist, in any quantity, on the moon. Internal heat, while undoubtedly present, is a "low grade source". It is diffuse, thus difficult to exploit. Hence the choice would seem to lie between (c) and (d).

2. It is advantageous to use solar power on the moon if at all possible. The energy is free, and there is no interfering atmosphere. Hopefully, transportation cost from earth would be less for a solar system than for a nuclear power system, allowing for local (moon-based) fabrication of many bulky items in the solar collector and the radiator. The lesser hazard constitutes a further advantage.

3. The usual disadvantage of solar power, as compared to nuclear power, is the intermittency resulting from the day-night cycle. This disadvantage would almost* vanish if the solar collector could be located at the crest of a peak near one of the lunar poles. Certain of these crests enjoy almost* perpetual sunlight. (7,8,10)

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*There would still be eclipses

4. From the above, it would appear that a solar installation is worth considering. There would still be a question how best to utilize the available solar energy.

5. With present technology, the outstanding possibilities are

   a. thermoelectric conversion
   b. photovoltaic conversion (solar cells)
   c. thermionic conversion
   d. dynamic heat engine cycles

      1. Stirling cycle
      2. Brayton cycle
      3. Rankine cycle

The direct conversion methods a,b, and c, are presently enjoying rapid development and improvement, particularly for spacecraft applications. They tend to be lightweight and compact. Broadly speaking, they have the advantage in the low power range, up to about 5 Kw. At higher power levels, dynamic heat engines show the better economy. Complete heat engine systems are complex and extensive, but this fact in itself is not necessarily a drawback, given a firm base of operations with plenty of space, and with available bulk raw materials. Moreover, heat engine technology is already well developed. A vast amount of engineering knowledge is immediately applicable. In all, the arguments in favor of the heat engine system seem to be decisive.

6. Choice amongst the 3 heat engine cycles will hinge largely on efficiency, and more particularly on power per unit weight. The Stirling cycle is the only one which matches the Carnot cycle in efficiency, but it has the drawback of utilizing reciprocating pistons in cylinders, not high speed turbines. Accordingly, in the power range above 25 Kw, it fails to compete favorably on a cost basis. Both the Rankine cycle and the Brayton cycle admit the use of turbines, and are therefore preferred in the power range presently of interest.

In recent years there has been much attention to the Brayton cycle for power plants operating under zero gravity. The working fluid remains in a gaseous state throughout the cycle, hence there is no problem of zero gravity boiling. In the absence of experimental data on boiling heat transfer under lunar gravity, one might assume that the situation on the moon would resemble the earth-based situation more closely than it does the zero gravity situation, and plan accordingly. On that basis, one would tend to prefer the Rankine cycle because of its inherently higher efficiency.

Actually, as will appear later, the analysis leads to a super-critical Rankine cycle, extending far into the superheat region. Such a cycle becomes, in effect, a sort of hybrid. In its heat absorbtion portion, it resembles the Brayton cycle; in the heat rejection portion, it is like an ordinary Rankine cycle with superheat. In any case, it seems best for now to look first at the Rankine cycle, bearing in mind that the Brayton cycle has not really been ruled out.

OPERATING TEMPERATURES

Solar energy is free, and is in constant supply. It can not be hoarded, nor turned off. Hence there is no direct reason for economizing in one's use of it. In a word, there is no incentive to achieve high efficiency for its own sake.

High efficiency remains a desirable attribute because it contributes to the achievement of high output per unit cost, as for example, by minimizing the size of the solar energy collector. Looking for ways to get high efficiency, one would normally try to arrange for a high temperature of heat absorption T_1 and a low temperature of heat rejection T_2.

In many situations, this broad classical principle has to be compromised, for one reason or another. (On earth, low temperature heat sinks are not readily available; in orbiting space vehicles, T_2 has to be kept fairly high so as to utilize a small, lightweight radiator structure.) At the lunar pole, such restrictions are greatly relaxed. The radiator can be placed in permanent shadow, allowing the rejection of heat to an almost black sky. This gives a low value for T_2, which in turn means a radiator of very large area. Such a thing would be prohibitive on a spacecraft; but on the moon, under our assumed conditions, it may be possible, even practicable.

Actually, for maximum ideal (Carnot) efficiency, the quantity to maximize is $(T_1 - T_2)/T_1$. Speaking generally, then, one can gain more by depressing T_2 than by elevating T_1. This fact can easily be lost sight of, since, commonly, there is a bottom limit, not very low, on T_2, and efficiency can be increased only by raising T_1. If, on the other hand, one could establish T_2 at an extremely low value, the inducement to increase T_1 would not be nearly so strong.

Absolute theoretical upper limit for T_1 is fixed by the temperature of the radiating surface of the sun ($\sim$ 5700 °K), and is far above any temperature which can be practically utilized. Values of T_1 in the range of 2500 to 2800 °R have been suggested[11] as being within reach by "reasonable extrapolation of current technology". Such temperatures, however, are well beyond those normally employed in earth-based power plants ($\sim$ 1000 °F), and have been suggested for space-station power plants only because the need to keep T_2 high makes it necessary to keep T_1 still higher. Provisionally, one might settle for a value of T_1 somewhere between 800 °K (rule-of-thumb for the usual steam power plant) and 1500 °K (approximate maximum considered for spacecraft systems utilizing alkali metal vapors).

It is easier to reach a decision about T_2. Effective temperature of the black sky, complete with stars, has been estimated at 30 °K. This temperature, however, is much too low for a practical radiator on the Moon's surface. Allowance should be made for the temperature of lunar crust. Watson, Murray, and Brown[12, 13] have chosen the figure of 120 °K as a reasonable upper limit for the temperature of lunar surface in perpetual shadow. This figure allows for the effect of heat flow from the lunar interior; also, for heat flowing laterally through the crust from warmed areas. Some further allowance should be made, certainly, for the warming effect of the power plant operation itself. We therefore select, tentatively, the figure of 150 °K. At this temperature, one could reject heat not only by radiation to a black sky, but also, perhaps, by radiation and/or conduction downward to the surface. Some modification of this figure can be expected, once a choice of working substance has been made, and calculations on the radiator carried through. For the moment, however, we note the prospect of very high Carnot efficiencies. With T_2 = 150 °K,

Efficiency = 81% (T_1 = 800 °K

= 90% (T_1 = 1500 °K

The above figures underline the point already made about ideal operating temperatures. If we can get 81% efficiency (ideal) at an easy-to-handle 800 °K, and the input energy is free, why struggle to work at the high level of 1500 °K to gain only a small percentage?

CHOICE OF WORKING FLUID

To make a start somewhere, one might scan the list of substances capable of going through a gas-to-liquid change of phase at 150 °K or thereabouts, at some reasonable pressure. Such common atmospheric gases as Nitrogen (crit. temp. 127 °K) and Argon (crit. temp. 151 °K) are too volatile. The usual refrigerants (NH_3, SO_2, the various freons) are not volatile enough. Among the rare gases, however, are two likely candidates:

Krypton (triple point, 116 °K, 548 Torr; critical point 209 °K, 54.3 atm)

Xenon (triple point 161 °K, 612 Torr; critical point 290 °K, 58 atm)

Aside from the fact that these gases condense at approximately the desired temperature and pressure, there are three strong arguments in their favor:

1. Both gases, being monatomic, show a high value for γ, the ratio of specific heats. This fact, in turn, means that a large temperature range corresponds to a relatively small pressure range.

2. Krypton and Xenon are chemically inert. This characteristic at once removes all need to worry about corrosion or chemical instability.

3. There is reason to believe that both gases occur, albeit in a state of extreme attenuation, in the lunar "atmosphere". While we are not aware of any practical means of collecting a gas existing at such extraordinarily low density, one can hardly rule out the possibility that a means may be invented.

Saturation curves for Krypton and Xenon appear in figures 1 and 2. (Original data from ref. 12). Neither gas is quite right for condensing at the preselected temperature. At 150 °K, the saturation pressure for Krypton is 6.4 atmospheres --- somewhat higher than optimum. For Xenon, on the other hand, 150 °K lies below the triple point. Using this gas, one would probably choose to condense at $\sim$170 °K (1.3 atmospheres).

Conceivably, one might contrive some advantage from working with a mixture of Kr and Xe. Such a fluid, of course, would not behave as a "pure substance"; each element would tend to act in accordance with its own partial pressure. Eventually, the possibility should be considered, if only because the mixture of gases might be easier to come by than either gas by itself. One may reasonably expect that if either element is present on the moon, the other will be also. Any scheme to concentrate such extremely diffuse (and chemically inert) material would probably yield as its end product a mixture which one might prefer, in the interest of economy, not to have to separate into constituents. Parenthetically, the possibility of using mixed noble gases in a Brayton cycle has already been considered, by authors[13] seeking to optimize the design of a space power plant in the 10-100 Kw range.

For the present, however, we put aside this line of thought, and consider the use of Xenon, with T_2 fixed at 170 ^{o}K. P_2, then, will be 1.3 atmospheres. If P_1 is established at 57.6 atmospheres (the critical pressure) then T_1, calculated by the perfect gas law, becomes approximately 775 ^{o}K. This figure lies comfortably within the range of practicality, and gives rise to a Carnot efficiency of 78%.

Actually, of course, the perfect gas law gives only a rough approximation for T_1. Vapor properties, derived experimentally, would furnish a much better base for calculation. But for Xenon, these properties appear to be available in the literature only through a small range of temperature[15].

With due allowance for the crudeness of this first approximation, and for a number of factors which have yet to be considered, one can draw two firm conclusions:

a. It would be feasible to use Xenon as the working fluid, over the entire temperature range from T_1 = 800 ^{o}K to T_2 = 170 ^{o}K. Pressures would be easily manageable throughout the cycle.

b. Xenon could be used in this way, and the cycle would still belong, technically, to the "Rankine cycle with superheat" classification; but the importance of the superheat portion would be far greater than usual. Indeed, during the heat input portion of the cycle the entire action takes place at constant pressure, not constant temperature, and the result is something more like the Brayton cycle than the Rankine cycle.

In consequency of item (b) above, the Xenon cycle would be considerably less efficient than a Carnot cycle between the same temperature limits. In figure 3B, the area IJKL, corresponding to heat added, is roughly trapezoidal. Comparison of area IJKL (using the trapezoidal approximation) with area of heat rejection, IJML, leads to an estimate of thermodynamic efficiency of 65%. Even this may be optimistic, for the area IJKM is not exactly a trapezoid. Undoubtedly, 60% would be a better figure.

COMPOSITE CYCLES

There is, of course, a classic solution to the problem of making the Rankine cycle approximate the Carnot cycle, when the temperature range is wide: the binary vapor system. In the present situation, it is natural to think of a binary vapor cycle, wherein Xenon would be the low temperature fluid. The high temperature fluid, ideally, should be one having a vapor pressure of some 25 to 75 atmospheres at a conservative maximum temperature ($\sim$ 800 °K) and about 1 atmosphere at a reasonable temperature for transfer of heat to Xenon, preferably somewhere below 290 °K, the Xenon critical. It quickly becomes evident, however, that no single fluid can meet these criteria. Mercury and the alkali metals satisfy the high temperature, but not the low temperature requirement. The reverse is true for NH_3, SO_2, and other common refrigerants.

Steam is a possibility, although the high critical pressure is a decided disadvantage. Among the many advantages are

a. Mature state of the art

b. Possibility of finding ice deposits at the lunar poles[18, 19]

Indeed, whether or not ice is found on the moon, it is obvious that a lunar colony must have water; if the power station can utilize a material which will necessarily be present anyway, the logistics and supply problems will become simpler by some amount. In fact, one might envision a scheme whereby the use of steam in the power plant is integrated with the "domestic" use of water by the lunar colony. The thermodynamic boil-expand-condense cycle becomes then the purification (distillation) cycle necessary for re-use of water by the life support system (just what a good power engineer would normally wish at all costs to avoid). To follow up on this possibility, one would of course have to design the boiler in a special way, to permit the continuous or short-term-periodic removal of the impurities in the feed water. It has been shown by Konikoff[20] that a partially closed-cycle physico-chemical ecology can be devised whereby an initial supply of water is not only maintained, but slowly increased, due to indirect recovery of the water present in foodstuffs, provided these latter are continuously supplied in open cycle fashion. Since a pioneer moon colony will probably need to utilize partially closed systems at first, before it becomes feasible to switch over to a fully closed system, there could be built up, during the early stages of lunar colonization, a reserve of water which might become available, at the proper time, for use in the electric power plant.

In view of all that has been said about free solar energy, and the reasonably high efficiency of the supercritical Xenon Rankine cycle, it may be somewhat academic to push further in the search for a composite cycle, employing different substances in such a way as to realize the classic Rankine cycle pattern on the temperature-entropy diagram. Such a search, indeed, leads to a quaternary vapor cycle. Table 1 affords a synoptic view of temperature and pressure relations which could hold for a 4-fluid system, utilizing mercury, water, ammonia, and xenon.

Solar energy would be used directly to boil the mercury, at moderate pressure and conservative temperature, also to superheat the Mercury slightly. The superheat serves to ameliorate the erosion problem[21]; moreover, as will be seen in the section on the section on the solar boiler, it is probably easier to design for superheat than for saturation. The superheat called for in Table 1 is very slight, thus avoiding the materials problems associated with high temperatures. One might prefer, however, to go to somewhat higher temperatures to prevent condensation of mercury within the turbine.

In either case the mercury exhausted from the turbine is condensed at a temperature convenient for boiling water. The action takes place in a heat exchanger wherein pressures on both sides are regulated to maintain an 11 °C temperature difference between condensing mercury and boiling water.

The resulting saturated steam is delivered to a separate solar furnace, where it acquires several hundred degrees of superheat. It then expands through a turbine to a temperature not far above that which will normally be maintained in the living spaces of the colony. Heat of condensing steam serves to boil the ammonia. Saturated NH_3 vapor takes on superheat, expands through a turbine to a temperature well below the critical for Xenon, where once again the condensation of the spent vapor boils the lower temperature liquid, and the final stage goes through the superheat, expansion, and condensation steps.

The system just described is indeed a complicated monstrosity, and certainly would not be adopted merely for the sake of conserving sun power (which is going to be wasted anyhow, in one way or another). The only real justification for considering the quaternary vapor cycle lies in the possibility that by its very complexity it may serve a number of auxilliary functions (temperature control, process heating, refrigeration, distillation, etc.) for the colony. A highly sophisticated control system might allow for both heating and cooling by diversion of the appropriate fluids, at the right places in the cycle. Such heating and cooling would constitute an efficient use of power resources, since the heat transfer would be always into a fluid which needed to pick up heat, or out of a fluid which needed to lose heat, in the normal process of operation of the cycle.

Finally, it may be remarked that the complexity of the quaternary system does not necessarily imply low reliability. On the contrary, its reliability could be greater than that of a single fluid system by the addition of a few relatively simple provisions for operation in alternate (degenerate) modes. Standby loops would need to be provided, such that the higher temperature fluids could, in emergency, flow through condensers radiating to space. Such an arrangement would cost very little additional because of the smaller radiator area needed for the hgiher temperature fluids. Similarly, provision would have to be made for utilization of the superheaters, in emergency, in place of the heat-exchange boilers. Sophisticated control would be necessary; but the principle is straightforward.

SOLAR COLLECTORS

General principles of solar collector design are well understood, and have been reviewed in recent literature.[17,22] Parabolic or paraboloidal reflectors would be preferable to lenses. The reflectors could be fabricated on the moon, of light weight material; lenses might have to be lifted from earth. Moreover, the pitting of lenses by meteorites could be a problem. Parabolic reflector surfaces would also be vulnerable to meteorite puncture; but in the case of these latter, the degradation rate would undoubtedly be slower, and the difficulties of repair less severe.

Mackay[17] has suggested an array of parabolic cylinders, each with its own boiler tube in the focal line. The entire array is caused to move as a unit, facing the sun; turbine, pump, radiator, and all accessories are mounted upon the single unified structure.

In the larger, more complex and more permanent installation presently under consideration, it would not be feasible to mount everything together. Instead, the design represented in figure 5 is suggested. This design would be appropriate to serve the heat-input portion of the Xenon Rankine cycle described earlier, and with some modification would be applicable to other cycles.

The boiler tubes stand vertically, in a bundle, at a point which is the center of symmetry in the plan view. These tubes would not have to rotate. The plumbing could be simple and permanent.

Closely surrounding the bundle of tubes there would be a cylindrical shell, to control radiation entering and leaving the tube assembly. This shell serves to block radiation from the tubes outward into space. There is, however, a relatively narrow slit running up and down the length of the shell. This slit permits entry of radiation from the mirror, and also, unavoidably, permits re-radiation outward from the tubes. The cylindrical shell must rotate with the apparent motion of the sun, the slit always facing directly away from the sun, toward the reflector.

The mirror would consist of highly reflective foil material in a layered construction, mounted upon a structural frame. The structural frame, in turn, would rest upon a circular track, upon which the entire assembly could roll. A simple optical sensor would govern the motion of the assembly, maintaining the axis of the parabolic reflector in constant alignment with the incoming rays from the sun.

Geometrically, the mirror would consist of two parts, the lower part being a parabolic cylinder and the upper part a matching paraboloid of revolution. Focal length would of course be the same for the two, and would be slightly less than the distance from the vertex of the parabola to the center of the boiler tube structure. The focal line (for the parabolic cylinder) and the focal point (for the paraboloid) would lie just inside the slit in the radiation shield which encloses the tube assembly.

In operation, the pressurized Xenon would be pumped in at the bottom of the tube structure and would slowly rise, expanding and picking up heat in the process. Since the concentration ratio is higher for the paraboloidal than for the parabolic reflector, the highest temperatures occur at the top. There would be a steady flow of heat from the top downward along the tubes, the heat also draining into the fluid, along the length of the tubes. The gas itself, upon reaching the top of the structure and acquiring there its maximum temperature, would then be forced to flow downward in a well insulated tube placed in the center of the bundle. Turbine is located underground, preferably close to the above described solar furnace assembly.

Suppose the dimensions of the mirror to be as follows:

20 meters, width of shadow
30 meters, height of cylindric elements
20 meters, diameter of rim of paraboloidal crown.

Total projected area, available for trapping sunlight, would then be 818 square meters, minus some small amount due to the shadow of the absorber. At 1394 watts/meter^2, this comes to $\sim$ 1140 kw gross energy input from the sun. Only a portion of this energy can be transferred to the working substance, depending initially on such factors as

C, the concentration ratio

$$= \frac{\text{radiant flux density received by absorber}}{\text{radiant flux density impinging on concentrator}}$$

N_c, the concentration efficiency

$$= \frac{\text{radiant energy entering slit}}{\text{radiant energy impinging on concentrator}}$$

and also, eventually, upon the temperature relationships existing between the absorber surfaces and the working fluid. The concentration ratio depends essentially upon the rim angle θ (73°, in our suggested design) and the width of the slit. A concentration ratio of 100 should be attainable, without undue difficulty, in that portion of the system wherein the focussing is two dimensional[22]. In the "crown" portion, with 3 dimensional focussing, one can obtain concentration ratios far higher ($> 15{,}000$), with correspondingly high temperatures, if necessary. It is desirable, however, not to go to extremely high temperatures --- not only on account of materials limitations, but also to keep the re-radiation losses to a reasonable level. Temperatures should be just high enough to make for effective heat transfer from tube walls to fluid.

In the proposed design, the cylindric part of the system, by itself, could operate at 660 °K, (with the working fluid draining away 80% of the available heat) or as high as 800 °K (with only 50% of the available heat being transferred to the fluid). Since in the proposed design there would be an extra-high temperature hot spot at the top of the assembly, temperatures down the length of the tube would be influenced accordingly. Exact analysis of system efficiency would best be accomplished by a computer, considering the varying temperatures and gas densities from point to point in the flow line. However, in view of the flexibility resulting from the combination of 2 dimensional and 3 dimensional focussing, it seems altogether reasonable to expect an overall efficiency (gain in enthalpy divided by solar input) not far below the maximum considered by Mackay,[17] say, about 80%. This leads to a figure of 910 Kw input to the thermodynamic cycle, which in turn would break down to something like

480	Kw	useful output
400	Kw	rejected in radiator
30	Kw	miscellaneous losses

Reviewing the above, one might raise the question of scaling either up or down in capacity rating. Especially might one be interested in the possibilities in the upward direction. Since the reflector has to be mounted upon a slowly rotating platform, there is indeed a practical upper limit to its size and weight, even after all allowances are made for low gravity and the absence of weather. The alternative of using a number of solar furnaces, each of moderate size, has drawbacks also. Two or more furnaces placed in proximity to one another would introduce the problem of shading. Yet they could not be scattered far apart, and stay essentially at the crest of a single peak. Conceivably, one may find a small, high crater, near the pole, with walls fairly even all around. Using such a formation, one might possibly mount small furnaces at intervals about the rim, tolerate a certain amount of shading, and use the floor of the crater for the radiator. Eventually, however, this question will have to wait; our present knowledge will take us just so far.

HEAT REJECTION

To radiate heat at 170 °K, at the rate of 400 Kw, would require an enormous emitting surface.

$$\text{Area} = 400(10^3) \Big/ \sigma \epsilon \left(T_2^4 - T_o^4\right).$$

With suitable choice of units, the radiator area somes out in square meters:

$T_2 = 170\ ^\circ\text{K}$

$T_o = 30\ ^\circ\text{K}$

$\sigma = 5.67(10^{-8})$ watts/(m^2) (°K)

$\epsilon = 0.95$ emissivity, dimensionless

Area = 8,900 square meters

To build a radiator covering 2 acres of territory in a lunar crater would indeed be a formidable undertaking, although not really far out of line with the cost and difficulties elsewhere in the proposed power project. One could, of course, cut the radiator area in half by working at 200 °K instead of 170 °K. Indeed, the practical answer may lie in this direction. Meanwhile, however, any possibility of a reasonable alternative should be explored.

In the absence of liquids on the moon, the prospects are not bright. If, however, it turns out that thermal conductivity of surface layers in the shadowed craters is higher than expected, one might unload some part of the heat by conduction. The heat thus transmitted to the ground would still have to radiate away, at the characteristically slow rate; but one could at least economize somewhat on the cost of tubes, fins, and working fluid to fill the tubes.

Other schemes can doubtless be thought of, whereby bulk material on the moon's surface can be caused to absorb heat at low temperature. For example, a mining operation might involve sifting or sieving of powdery or granular material. Just possibly, the temperature differential of 50 degrees or better could be turned to account, in those situations where the material would have to flow in a trough or conveyor channel. The possibility of finding ice deposits in shadowed areas has already been mentioned. Either alone or in combination with other materials the ice might be caused to pick up heat from condensing xenon, as a preliminary to other steps in transporation or processing. Once again, we seem to have run onto a problem where little headway can be made until people have landed on the moon and begun the work of exploration.

RECAPITULATION

Main conclusions are summarized below:

1. A physical situation unique in the solar system is found in the rugged terrain of the lunar polar regions.

2. This special situation prompts the consideration of a special kind of electric power generating station, such as would be impossible elsewhere.

3. High theoretical efficiencies can result from the immediate proximity of perpetual sunlight and perpetual shadow; but the endeavor to capitalize on this resource leads to serious practical difficulties in the design of collectors and radiators.

4. Xenon has properties which make it a very desirable working fluid for a heat-engine cycle, especially if the lower end of the available temperature spectrum is to be exploited. Other rare gases, alone or in combination, should also be considered.

5. Design of the special power station can and should be strongly influenced by the possibility of integrating the power requirement with other requirements, ecological and industrial, of the colony.

6. The search for feasible solutions to design problems leads at once into areas wherein present knowledge is grossly insufficient. Anything like a design optimization effort lies quite a few years ahead.

NOTES AND ACKNOWLEDGEMENTS

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Table 1
Quaternary Vapor System

Stage	Fluid	T_B	T_S	P_1	T_2	P_2
First	Hg	810 (998)	825 (1025)	12 (176.5)	500 (440)	0.052 (0.77)
Second	H_2O	489 (420)	800 (980)	21 (309)	311 (100)	0.0645 (0.95)
Third	NH_3	300 (80)	433 (320)	10.4 (153)	233 (-40)	0.71 (10.4)
Fourth	Xe	222 (-60)	350 (170)	10.8 (159)	170 (-154)	1.3 (19.1)

Temperatures are in degrees K, first line, and in degrees F, (second line).

Pressures are in standard atmospheres, first line, and in p.s.i.a., (second lin

T_B = boiling temperature

T_S = superheat temperature

T_2 = exhaust (condensation) temperature

P_1 = inlet pressure

P_2 = exhaust pressure

TABLE 1 Quaternary Vapor Cycle

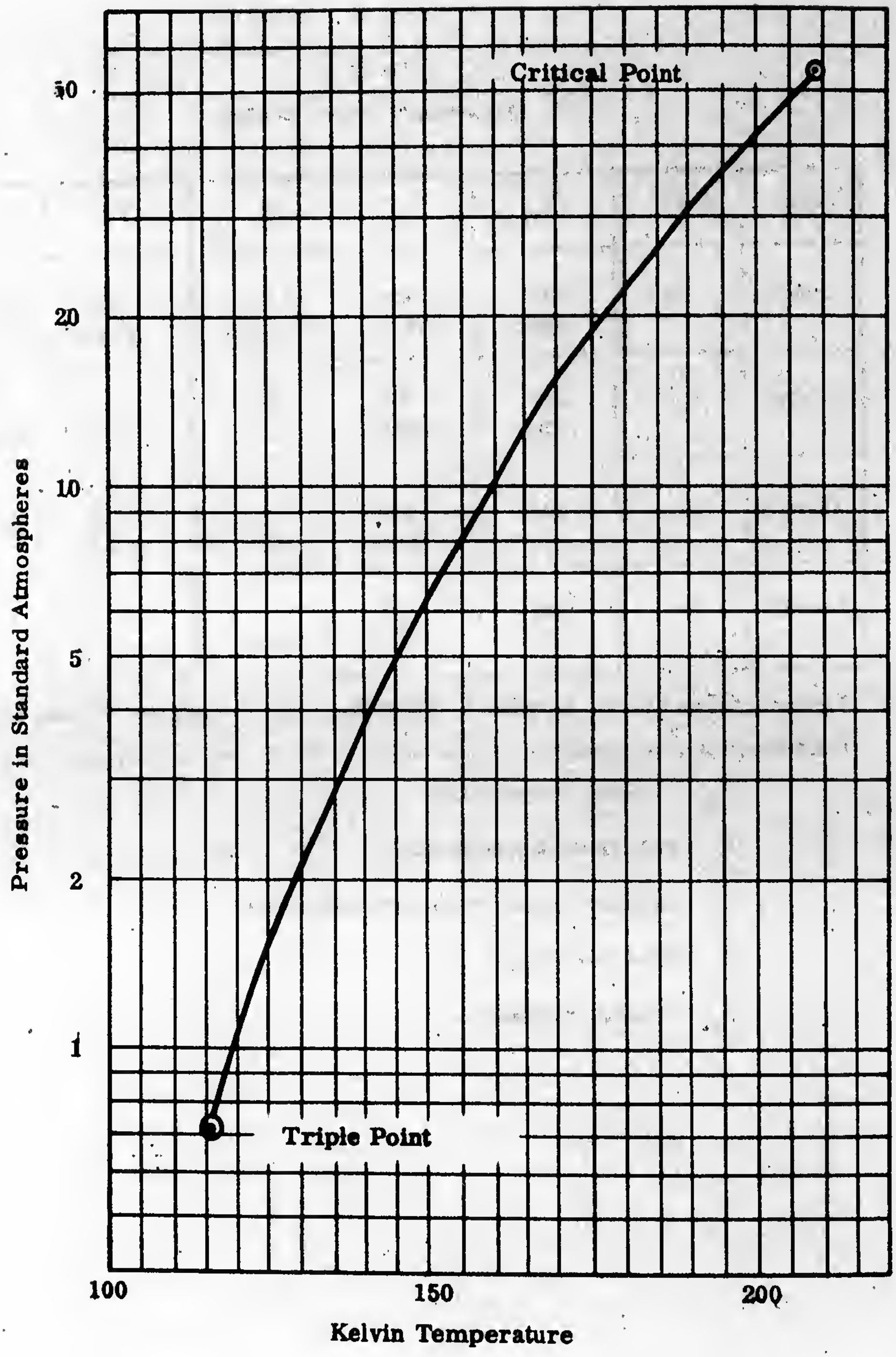


FIGURE 1 Saturated Liquid Line for Krypton

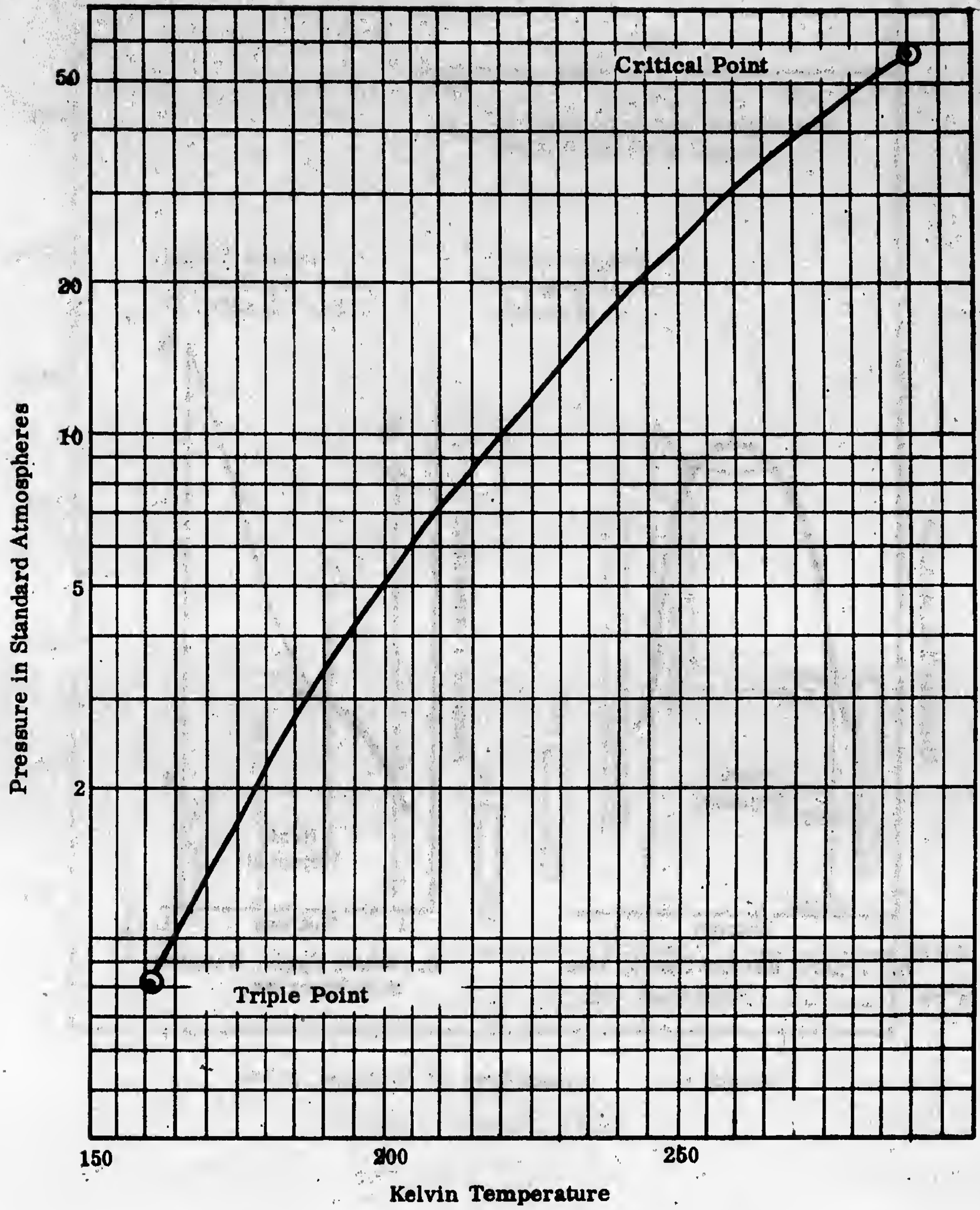
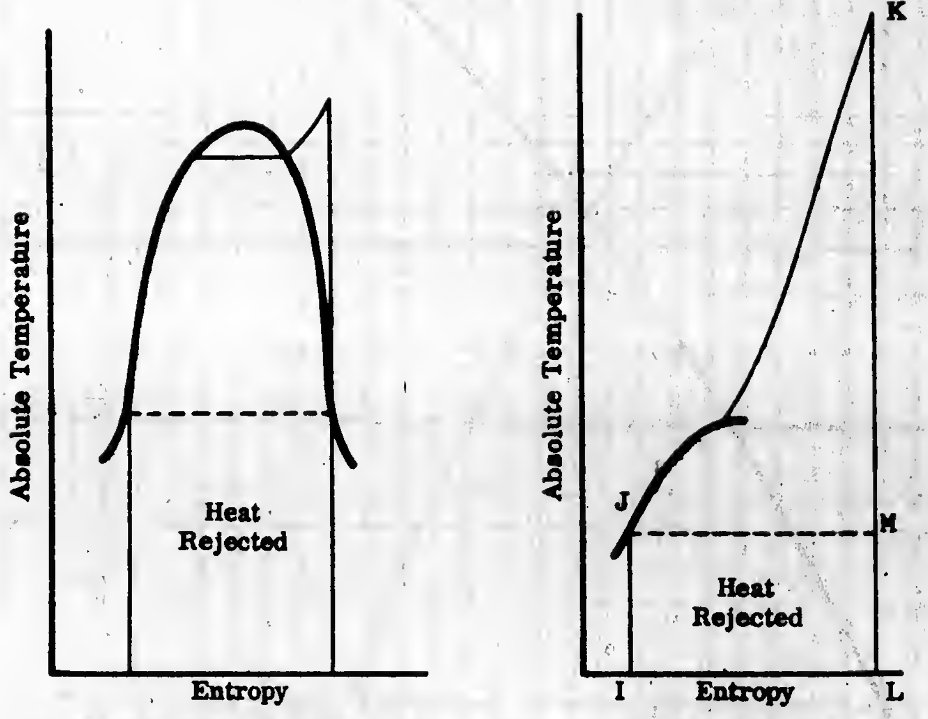


FIGURE 2 Saturated Liquid Line for Xenon



A. Typical Rankine Cycle, Ideal

B. Xenon Cycle, Rankine with Superheat

FIGURE 3 Comparison of Rankine Cycles

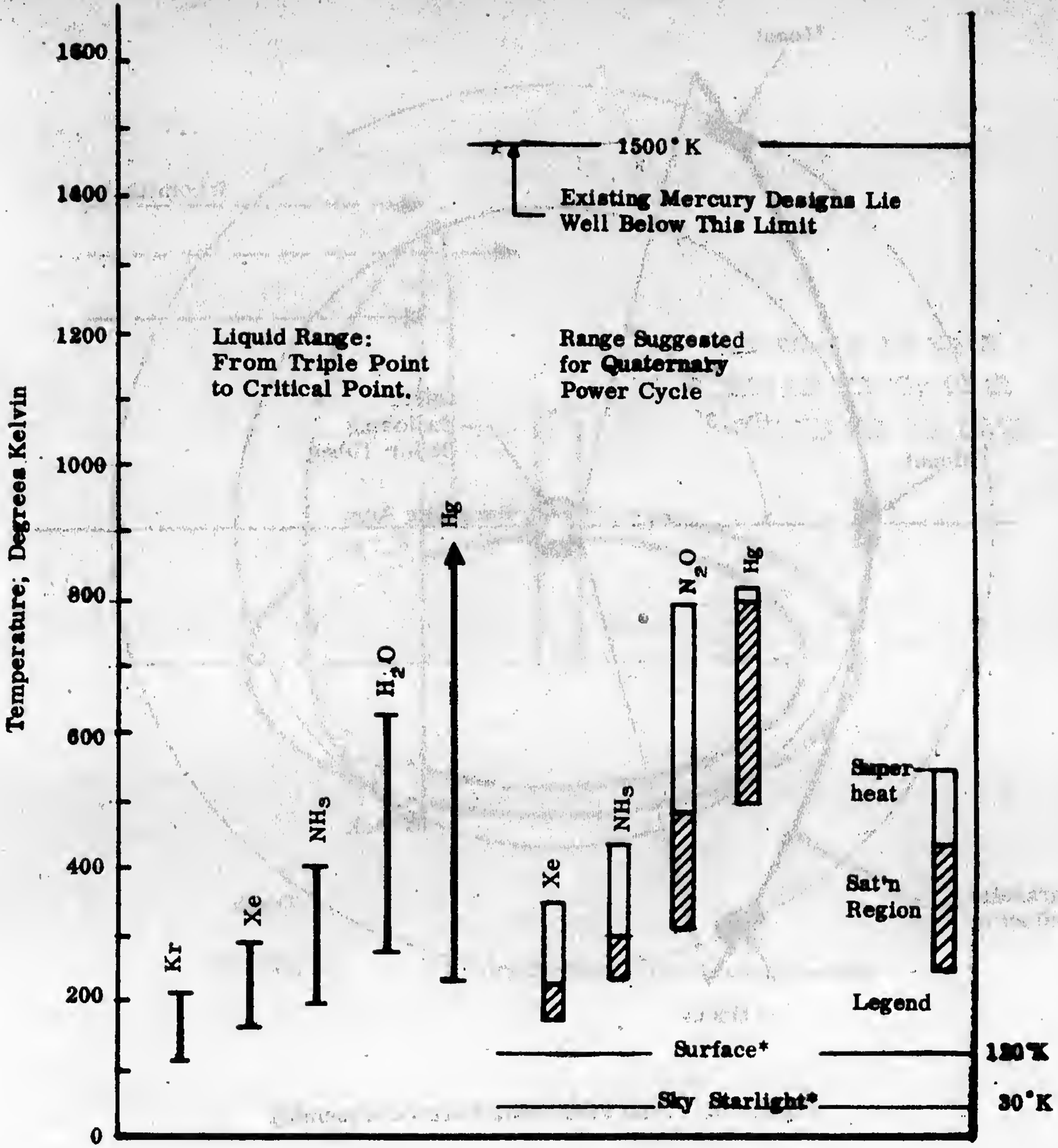


Figure 4. Working Fluids

* Lunar Temperature in Perpetual Shadow

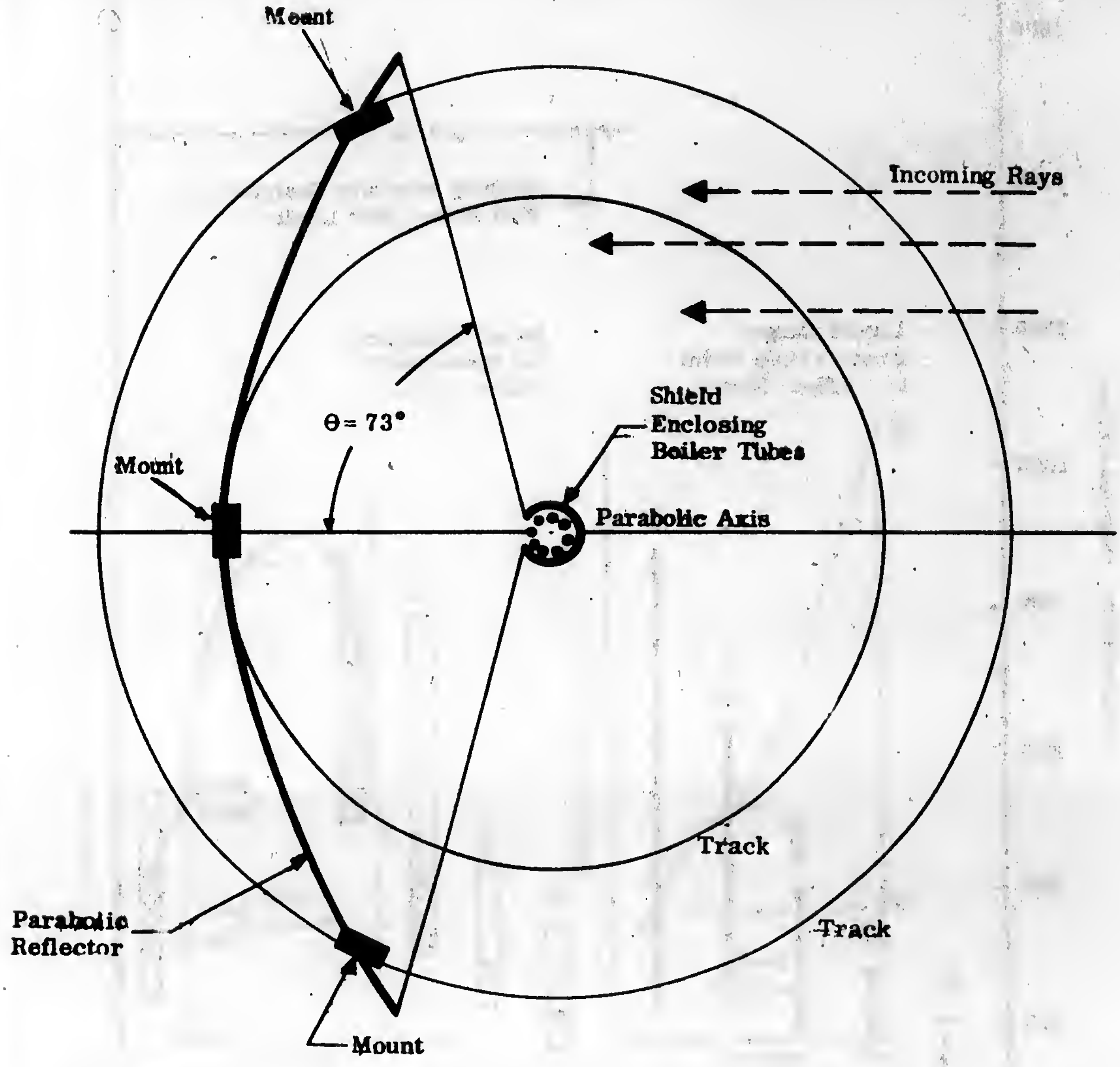


Figure 5A. Plan View Solar Furnace Assembly

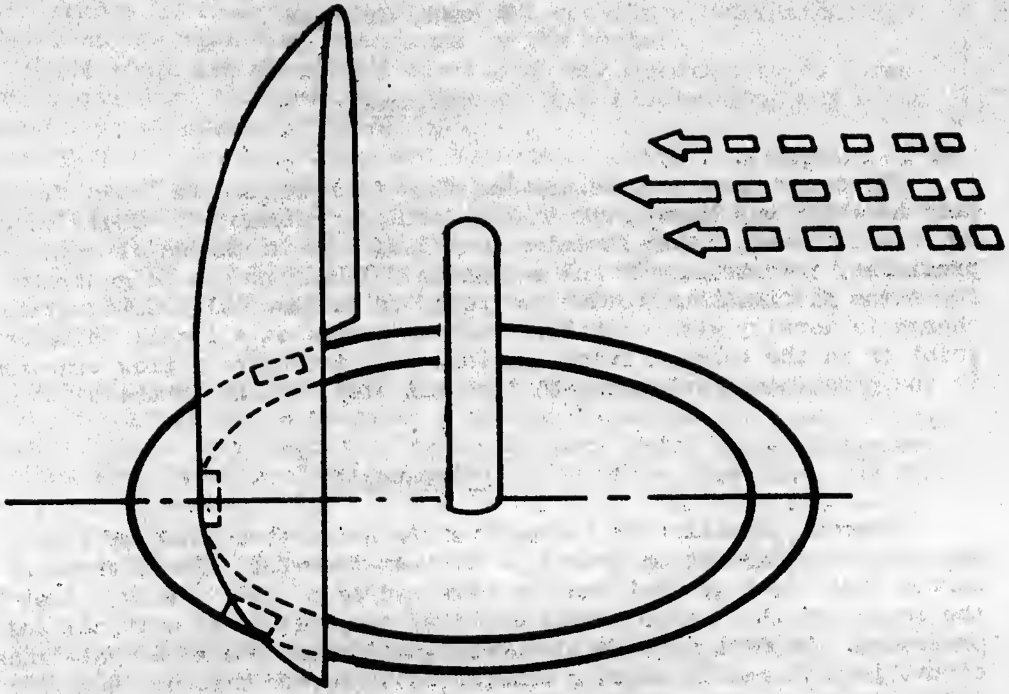


FIGURE 5B Pictorial View, Mirror and Absorber

ANALYSIS OF PRESSURE AND FLOW TRANSIENTS IN SYSTEMS WITH HEAT ADDITION

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Abstract

The occurrence of pressure and flow fluctuations in fluid flow systems with heat addition has been noted in many isolated instances. Under certain conditions of local boiling, which includes a boiling-like mechanism at super-critical pressures, intense sounds and mechanical vibrations can be generated. A mechanism for these fluctuations is derived based on fundamental considerations. A large change in density with enthalpy, such as occurs at a liquid to vapor phase change point or in the super-critical region, will introduce a flow transient. Changes in the frictional resistance to flow can also initiate a transient condition.

Introduction

Interest in utilizing convective regenerative heating of liquid rocket fuels has focused attention on this type of heat-transfer. For several of the hydrocarbon fuels such as RP-1 and for para-hydrogen, this involves heat-transfer in the region of the thermodynamic critical temperature at critical and supercritical pressures. Reports of several investigations to determine heat-transfer coefficients in this region contain references to severe pressure and flow oscillations under certain operating conditions. Attempts to explain these phenomena have not been successful to date. In this paper is proposed the mechanism for the pressure and flow oscillations based on a one-dimensional model of the diabatic flow system.

Two cases where heat transfer to fluids in the thermodynamic critical region is encountered are: the operation of a fluid flow system, particularly a natural-circulation system near the critical point, is desirable because of the large heat capacity and large density gradients that exist in this region; and, the fluid passes through the critical region as it is being heated in a super-critical pressure system. An example of the second case is the convective regenerative heating of liquid propellants at the rocket nozzle such as the RP-1 fuel mixture or hydrogen.

The occurrence of pressure fluctuations severe enough to induce mechanical vibration of the tube and subsequent rupture has been reported in several papers. Hines and Wolf[1] in a study of the pressure fluctuations encountered in a diabatic flow of RP-1 and Diethylcyclohexane (DECH) at supercritical pressures reported fluctuations in pressure of 380 psi peak to peak with frequencies in the range from 1000 to 10,000 cps. The vibration was very destructive to thin-walled tubing. In tests with thick-walled tubing, similar pressure fluctuations were measured but the tubes were able to resist the effect of the mechanical stresses. Their paper mentions several similar occurrences reported in the literature.

In a documentation of boiling songs and associated mechanical vibrations of water systems with heat addition, Firstenberg[2] reported the results of informal interviews with various researchers in which boiling songs, mechanical vibrations, and pressure fluctuations were discussed. Several of the conclusions reached were:

1. Under certain conditions of local boiling, which includes a boiling-like mechanism at supercritical pressures, intense sounds (boiling songs) and mechanical vibrations can be generated.
2. The occurrence of these intense sounds and mechanical vibrations are intimately related to the operating conditions in the heater.
3. The boiling songs and associated vibrations are accompanied by large-amplitude pressure and flow fluctuations, although such fluctuations may occur without the generation of sounds or vibrations.
4. The occurrence of boiling songs and associated vibrations appears to be confined to local (subcooled) boiling systems. There has not been a recorded occurrence of these phenomena in bulk boiling (net vapor generation) systems.
5. The unusual phenomena result from pressure waves and/or flow fluctuations which induce variable heat transfer in the system. The variability in the heat-transfer rate and energy dissipation of the pressure wave results in the intense sounds and vibrations.

In the summary it was concluded that the severity of the vibrations and intensity of the sounds indicate a possible limiting condition for some boiling systems. Because of other more immediate needs in cases where these phenomena were observed, they were considered to constitute a nuisance condition, and thus, were not investigated systematically. It was recommended that some systematic investigation of these phenomena be initiated, directed at obtaining an understanding of the cause and effect relations, and discovering methods for eliminating the occurrences.

Goldman[3] in a report of heat-transfer experiments to supercritical water at 5000 psia distinguished between heat transfer without "whistle" or a "normal" mode and heat transfer with "whistle". The frequency of the whistle sound varied from about 1400 cps to 2200 cps. Inlet temperature appeared to have the greatest effect on frequency. Pressure had little, if any, effect on the frequency. A system pressure increase of about 50 psi occurred at the onset of the whistle sound.

Morgan and Brody[4] discuss the term "geysering" as applied to missile applications. The term "geysering" is applied to the specific phenomenon that occurs in a liquid system when a column of liquid in long vertical lines is expelled by the release of vapor at a rate in excess of that rate which may occur as a normal function of bubble release. This is also known as slug flow or the "coffee percolator" effect.

The result is an expulsion of liquid from the vertical line. Refilling the line by gravity action from a storage tank above the [illegible] results in a pressure surge analogous to water hammer. The pressure surges so produced can be very large and damage to feed lines, valve supports, and line supports as result of geysering is a common occurrence.

An effective solution to this problem was to run two lines from the liquid tank to the engine with an interconnecting line at the lower end. When one of the lines is insulated, a cold leg results and a natural-circulation is established. The resulting flow was sufficient to prevent the heat input in the uninsulated leg from vaporizing the fluid.

Flow and pressure oscillations in a liquid-oxygen system of the Saturn Booster rocket have been reported by Platt and Wood[5]. The oscillations occurred in the initial startup of an evaporator designed to supply gaseous oxygen to the fuel tank for pressurization by evaporating liquid oxygen. Pressure oscillated by as much as ± 300 psi at a frequency of 1/3 cps with the flow stopping in the worst cases. Amplitude diminished with time to ± 50 psi in the first 30 to 60 sec of the test run.

The occurrence of pressure and flow fluctuations in the cool-down of a cryogenic system is discussed by Bronson, et al[6]. Cool-down time was defined as the time interval between introduction of liquid hydrogen into a warm cryogenic system and the appearance of a continuous flow of liquid at the discharge end of the system. During experiments to measure the cool-down time, the pressure was found to undergo periodic fluctuations with a variable frequency. A model was presented to calculate the frequency of the pressure fluctuations.

In the first test run of a KIWI-B1 reactor at power with liquid-hydrogen propellant, severe hydrogen pressure fluctuations were observed at the pump outlet early in the startup cycle[7]. It was believed that these fluctuations resulted from an unchilled by-pass line at the pump outlet.

Lewis, Goodykoontz, and Kline[8] in a study of boiling heat transfer to liquid hydrogen report that under some operating conditions fluctuations of pressure and flow become uncontrolled.

This paper presents a hypothesis on the nature of the driving mechanism of the pressure and flow oscillations based on deductions from the conservation equations and the equation of state. It will be shown that the relationship of thermodynamic properties in the critical region or at a phase change point are very conducive to flow and pressure oscillations. The analysis is valid for either forced or natural-circulation flow. Plots of the similar thermodynamic characteristics of water and Freon-114 are presented.

Analysis

The flow system to be considered consists of a constant area tube through which a fluid is flowing either by forced or natural-circulation. A one-dimensional approximation is made. Laws of conservation of mass, momentum, and energy are written which describe the fluid flow through such a tube. The density is evaluated as a function of enthalpy alone at some reference pressure. Two expressions are obtained which describe the mass velocity in this system. One expression describes the local change in mass velocity as a function of the heat input and thermodynamic condition of the fluid. The second expression describes the time variation of the mass velocity produced by the interaction of the pressure drop with the local change in mass velocity. It is shown that the thermodynamic condition of the fluid conducive to large local changes in mass velocity with heat addition are those encountered when the fluid is in certain portions of the supercritical region or at a liquid to vapor phase change point.

The conservation equations for this system[9] can be written

$$\partial \rho/\partial t + \partial G/\partial z = 0 \qquad (1)$$

$$(\partial G/\partial t) + \frac{\partial}{\partial z}(G^2/\rho) = -\partial p/\partial z - \frac{f\,G\,G}{2d\rho} - \rho g, \qquad (2)$$

and

$$\rho(\partial h/\partial t) + G(\partial h/\partial z) = Q/A + \left[\partial p/\partial t + \frac{G}{\rho}(\partial p/\partial z)\right] + \frac{f\ |G|\ G^2}{2d\ \ \rho^2} \qquad (3)$$

An equation of state of the form

$$\rho = \rho(h,p^*). \qquad (4)$$

is used. This Equation states that the density can be evaluated as a function of enthalpy alone at some reference pressure and its validity depends on the major effect of heat transfer being a change in enthalpy. This simplifies the analysis considerably without eliminating the basic driving mechanism of the flow oscillation and pressure fluctuations.

Neglecting the effects on enthalpy rise rates of dissipation and pressure changes with space and time, Eq. (3) can be written

$$\rho(\partial h/\partial t) + G(\partial h/\partial z) = Q/A \qquad (5)$$

In the equation of state density is not a function of pressure, hence, this model does not permit compression with increases in local pressure. This means that local pressure and velocity disturbances are propagated at the speed of sound evaluated at the reference pressure of the fluid. A physical model is a string of beads moving along at some uniform velocity. If suddenly one of these beads expands to twice its linear length, all of the beads are accelerated along the length of the string. The momentum equation must therefore be integrated over the entire tube length and is used to represent the behavior of the average mass velocity, $\overline{G}$, only:

$$(d\overline{G}/dt) = (1/L)\ (\Delta P - F) \qquad (6)$$

where

$$\overline{G} = (1/L) \int_0^L G dz \qquad (7)$$

ΔP = available driving pressure and the total resistance to fluid flow, F, is given by

$$F = \left(\frac{G^2}{\rho}\right)_{exit} - \left(\frac{G^2}{\rho}\right)_{inlet} + \int_0^L \frac{f\ \ G\ \ G}{d\ \ \rho} dz + \int_0^L \rho g dz \qquad (8)$$

The mass velocity, G, is a function of z and t. Therefore, it is desirable to obtain explicit expressions for these relationships. Combining the equation of state and the continuity equation, Eq. (4) and (1), we obtain

$$(\partial\rho/\partial t) = (d\rho/dh)\ (\partial h/\partial t) = -\ (\partial G/\partial z) \qquad (9)$$

Eliminating $(\partial h/\partial t)$ in Eq. (5) and (9) gives

$$(\partial G/\partial z) = (1/\rho)\ (d\rho/dh) \quad G(\partial h/\partial z) - Q/A \qquad (10)$$

The momentum equation, Eq. (6), is repeated here for continuity of thought in the succeeding discussions

$$(d\bar{G}/dt) = (1/L)\,(\Delta P - F) \qquad (11)$$

The simultaneous solution of Eqs. (10) and (11) would yield the average mass velocity in the fluid flow system as a function of time. It is possible, however, to obtain a quantative picture of the flow behavior by examination of Eqs. (10) and (11).

Since the driving force for the flow and pressure oscillations in the system being considered is the heat input, we first examine Eq. (10). The term $d\rho/dh$ will always be negative for the fluids of interest in this type of heat transfer. In addition the numerical magnitude $(d\rho/dh)$ can be quite large under certain thermodynamic conditions to be discussed below. Thus we see that if the bracket term $[G(\partial h/\partial z) - Q/A]$ is other than zero, there will be a local change in mass velocity which will interact with the momentum equation, Eq. (11), to give a time rate of change in the average mass velocity.

It is of interest to inquire under what conditions the bracket term will be positive, zero or negative. In steady state conditions the heat input is absorbed by the flowing fluid, hence $[G(\partial h/\partial z) - Q/A] = 0$. If a change in Q occurs, the equilibrium balance will be upset. If the heat capacity of the fluid undergoes a change, then $\partial h/\partial z$ will change also. Hence we see the possibility of $[G(\partial h/\partial z) - Q/A]$ being positive, negative, or zero and if this difference is multiplied by a large number, quite large positive and negative values of $(\partial G/\partial z)$ can be obtained.

The momentum equation contains the damping terms which will determine the magnitude of the flow oscillations. With sufficient damping the flow oscillations could be suppressed but this is not a desirable operating condition.

In the thermodynamic critical region or at a liquid to vapor phase change point, the conditions favorable for the postulated oscillation mechanism are present. Scarcity of thermodynamic data in this region hamper extension to fluids other than those to be discussed. This, however, should point the way to the data that must be available for an analysis of flow instability.

One of the conditions necessary for large changes in $(\partial G/\partial z)$ is a large $(d\rho/dh)$. Plots of ρ vs h for Freon-114 and water are shown in Figures (1) and (2). As can be seen there is a relatively narrow range of enthalpies for which this condition will hold. Since a fluid being heated in this region would enter the heated section at an enthalpy below that for which $(d\rho/dh)$ becomes large, passing into the range for which $(d\rho/dh)$ is large, means that a fluid control volume experiences non-uniform accelerations. As the fluid enters the heater section, it experiences a uniform acceleration up to the point where $(d\rho/dh)$ changes slope. It then experiences a greater acceleration which means an increase in G and a decrease in $\partial h/\partial z$. If the $G(\partial h/\partial z)$ product increases, then $[G(\partial h/\partial z) - Q/A]$ changes in value resulting in a change in $(\partial G/\partial z)$. If the damping in the momentum equation is not sufficiently large, there will be sustained flow oscillations as $[G(\partial h/\partial z) - Q/A)]$ changes its value and sign periodically.

Another condition which is encountered is the change in $(\partial h/\partial z)$ which is due to a large change in heat capacity of the fluid in the critical region. This term can be determined from a measured temperature distribution between the inlet and outlet of the heater section as is shown below:

$$\partial h/\partial z = (\partial h/\partial T)_p (dT/dz)$$

As $C_p = (\partial h/\partial T)_p$ increases, (dT/dz) will decrease so that the true numerical magnitude trend of $(\partial h/\partial z)$ may be in doubt. Generally an increase in G would result in a decrease in $(\partial h/\partial z)$ but if the fluid is in the critical region, the decrease in $(\partial h/\partial z)$ may be offset as conjectured previously. All that is necessary to precipitate an unstable condition is that the product of $G(\partial h/\partial z)$ change.

Conclusions

It has been hypothesized, based on an examination of the basic conservation equations and the behavior of the thermodynamic properties of a fluid near its critical state or at a liquid to vapor phase change point, that the mechanism for the sustained pressure and flow oscillation is due to the fluid condition which can be defined in terms of a change in slope of $d\rho/dh$. It is shown that the change in gradient of this slope is the primary mechanism for the oscillation. It is conjectured that heat addition to any fluid which exhibits this characteristic would result in flow oscillations provided that the sharp change in the $(d\rho/dh)$ gradient occur somewhere in the heated section. Secondary mechanisms such as the change in heat capacity and heat transfer characteristics also enter the picture as shown in the combined state, energy, and continuity equation. The fact that viscosity is approaching or at a minimum concurrently with the primary mechanism provides decreased damping for the flow oscillations through the coupling with the momentum equation.

Further experimental work needs to be done to verify and refine this analysis. A heat-transfer loop presently being constructed at the University of Oklahoma will be used to check the preceding analysis. The oscillation threshold of the system with four different fluids is to be checked for agreement with the analysis.

Nomenclature

A	Cross sectional area for flow, ft^2
F	Total resistance to fluid flow -- composed of friction, elevation, and acceleration pressure loss, lbf/ft^2
f	Friction factor
G	Mass velocity, lbm/sec ft^2
$\bar{G}$	Average mass velocity in tube, lbm/sec ft^2
g	acceleration of gravity, ft/sec^2
h	Enthalpy, Btu/lbm
L	Tube length, ft.
p	Static pressure, psi
p*	Reference pressure for fluid property evaluation
Q	Linear rate of heat input to fluid, Btu/(sec)(ft)
t	Time, sec.
z	linear dimension. ft
ρ	Density.lbm/ft^3

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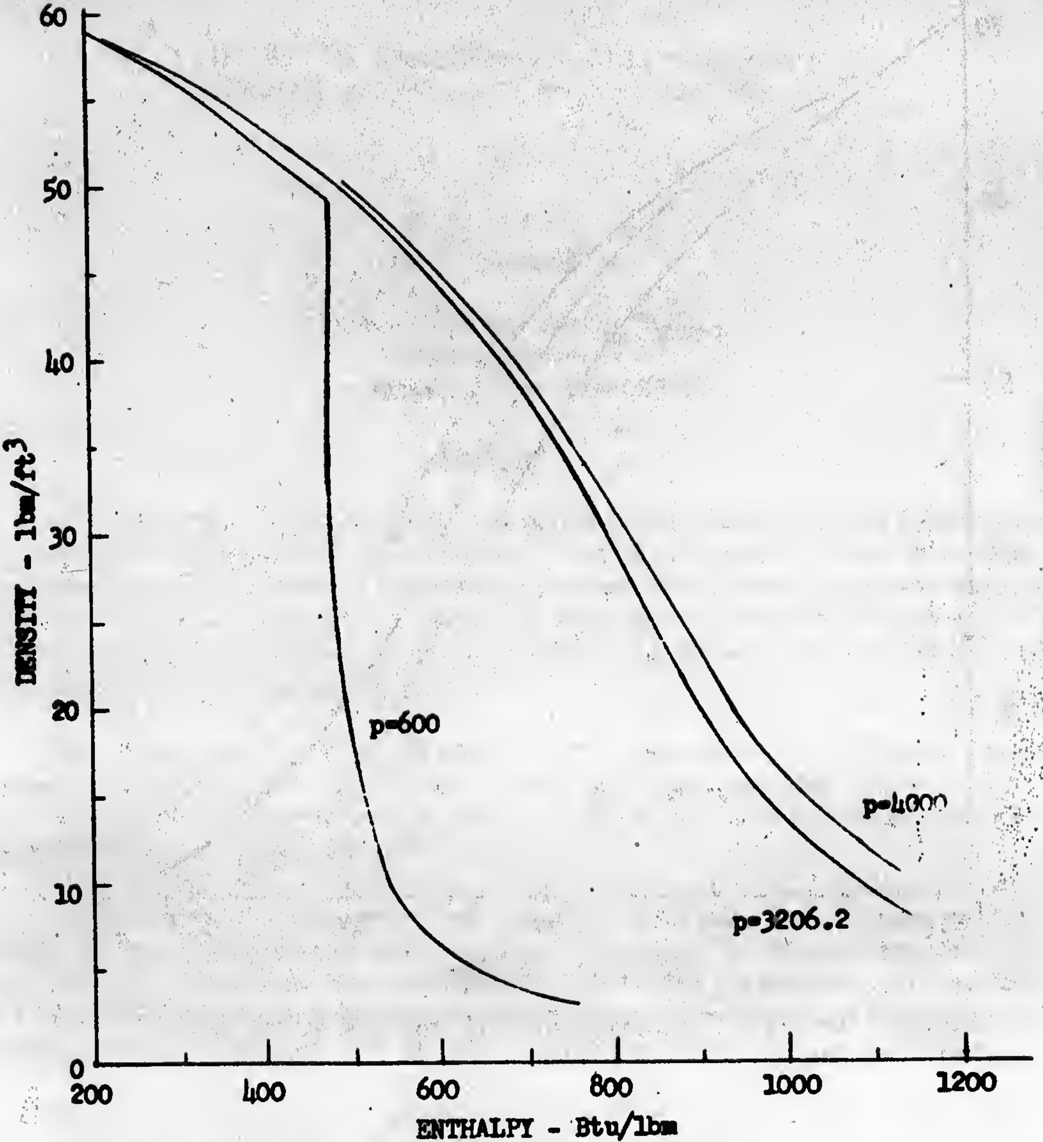

Fig. 1. Isobars of Density-Enthalpy for Water

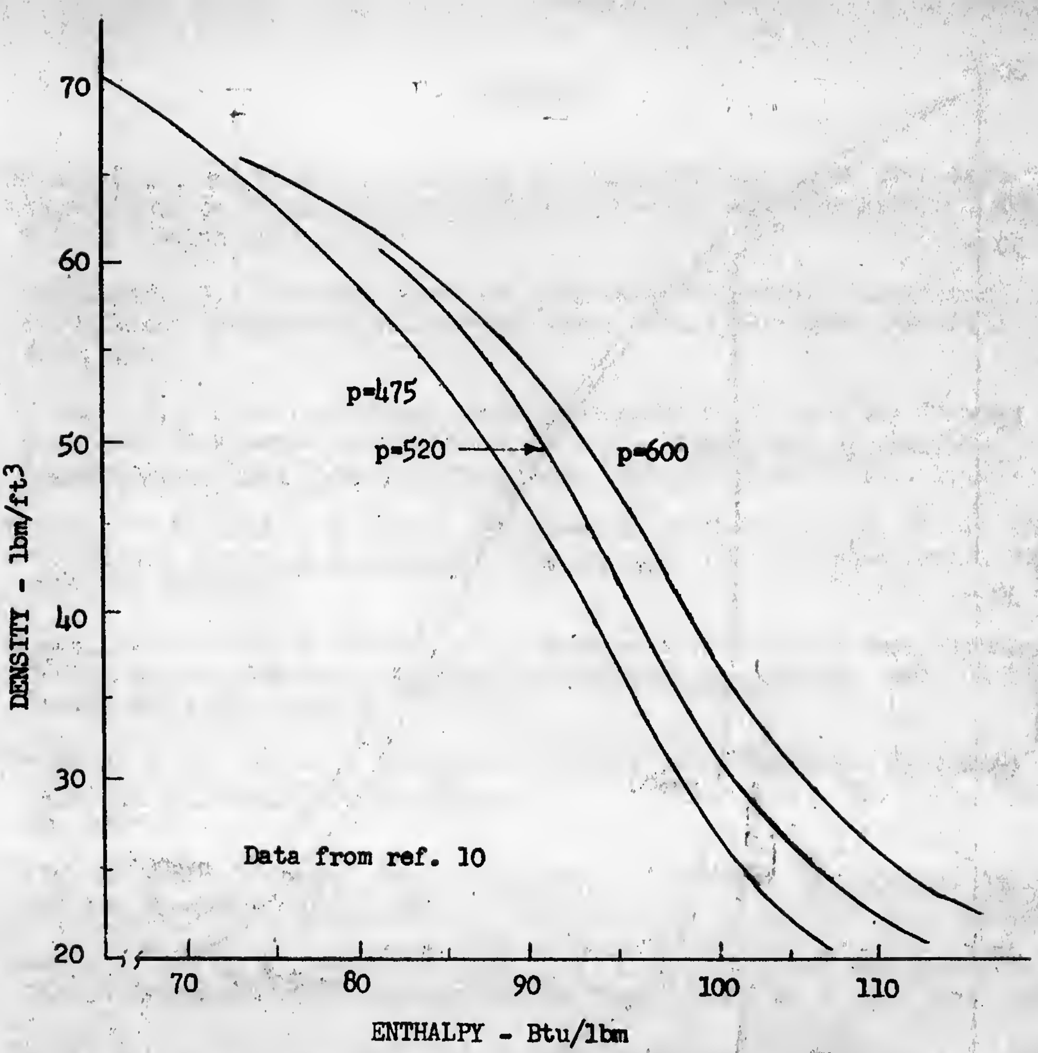

Fig. 2. Isobars of Density-Enthalpy for Freon-114 in the Critical Region

A METHOD OF EMBEDDING ACCELEROMETERS IN SOLID PROPELLANT ROCKET MOTORS

by

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Abstract

The USAF MINUTEMAN ICBM Transportation and Handling Test Program involves vibration testing of full scale motors. A major problem in vibration testing of large solid propellant motors is developing a method of embedding accelerometers in the viscoelastic propellant. The accelerometers are needed to determine experimentally the dynamic response of the propellant during the environmental tests, and to verify the motor's dynamic analysis.

The engineering analyses and experimental tests employed by Thiokol Chemical Corporation to develop the accelerometer installation method is the subject of this paper. The results of the instrumentation, and the degree to which the desired results were achieved, are discussed.

Extensive engineering studies were made to determine the minimum number, location, and type of accelerometers required to describe the dynamic response of the propellant. A technique was developed for controlling the position and orientation of the accelerometers when embedded in the motor propellant. A laboratory research program was conducted to develop reliable bond joints.

The configuration finally selected uses 24 triaxial accelerometers, of which 14 are embedded in the propellant and 10 are mounted on the propellant surface. The accelerometer and cable assemblies are precast in propellant wedges, which are bonded in the case prior to casting the motor. Piezoelectric accelerometers were selected in order to meet the dynamic response, safety, density; and temperature specifications.

As a final verification of the design concept, and to check out special manufacturing techniques and tooling, a subscale motor was successfully cast with accelerometers embedded in the propellant. This was accomplished prior to casting a full scale motor. Both the subscale and full scale motors were used in the vibration testing program.

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Thiokol Chemical Corporation utilized a new vibration facility for performing these vibration tests. This facility is equipped with electrodynamic and electrohydraulic exciter systems that have combined capabilities of sinusoidal force outputs from 0 to 200,000 vector force pounds, and frequency capabilities from DC to 2,000 cycles per second.

With the design concept developed for internally instrumenting solid propellant rocket motors, it is technically feasible to static fire these motors after the vibration tests are completed as final confirmation of the motor integrity. This capability was a secondary program objective.

Introduction

In the initial stage of the MINUTEMAN development program, the capability of the Stage I MINUTEMAN motor to withstand a prolonged transportation and handling environment was unknown. The chance that transportation and handling problems might occur as a result of vibration, shock, structural discontinuities, and localized loading required that the effect on reliability be investigated. In simple beams, these problems normally lend themselves to analytical study and analysis.

The Stage I motor, however, is a highly complex structure comprised of a thin-steel outer shell approximately 5 ft in diameter and 20 ft long, with a thick bonding liner. Key areas inside the case are insulated with a rigid insulating material bonded to the shell. The case is filled with a large mass of viscoelastic propellant having a star-shaped core chamber through the center. With such a composite structure, the mechanical properties of which are either unknown or vary from specimen-to-specimen, classical beam analysis becomes inadequate in describing the response of the motor to a dynamic input.

Because of the specialized nature of this problem, a consulting service in Applied Mechanics performed an analysis and predicted the dynamic response of the Stage I motor when subjected to transportation and handling environments.

To verify this analysis by tests, Thiokol Chemical Corporation conducted road, air shipment, and obstacle course tests to determine how well the motor would withstand actual field conditions. Although a great amount of useful information was obtained from these tests, no satisfactory means was found to control the test environment over the entire frequency range and at the acceleration levels required to check the analytical predictions or to determine the dynamic response of the motor. The frequency excitation levels were so low in value that the motors tested by these methods responded essentially as a rigid body. No fundamental frequencies or mode shapes were measurably excited.

To determine the fundamental frequencies, mode shapes, and propellant response, and to check the analytical predictions, Thiokol designed and constructed a vibration test facility at the Wasatch Division with the capability of testing the full scale Stage I MINUTEMAN motor.

To fully describe the behavior of the motor, the propellant response was required as well as the case response. Engineering studies were conducted to establish the minimum number and location of accelerometers required to determine the dynamic response of the propellant during the vibration tests. A method of embedding these accelerometers was developed that enables the accelerometers to be fixed in prescribed locations during propellant casting. No adverse dynamic effects occurred on the motor or propellant from the embedded accelerometers and cable assembles.

Discussion

One of the major problems associated with this program was trying to determine the number and corresponding locations of accelerometers required to describe the dynamic behavior of the propellant during vibration tests.

Joint studies by Space Technology Laboratories, Dyna/Structures, Inc. and Thiokol indicated that the dynamic response of the Stage I propellant grain could best be determined by embedding three rows of accelerometers in the propellant at each of six linear cross-sectional stations within the motor (Figure 1).

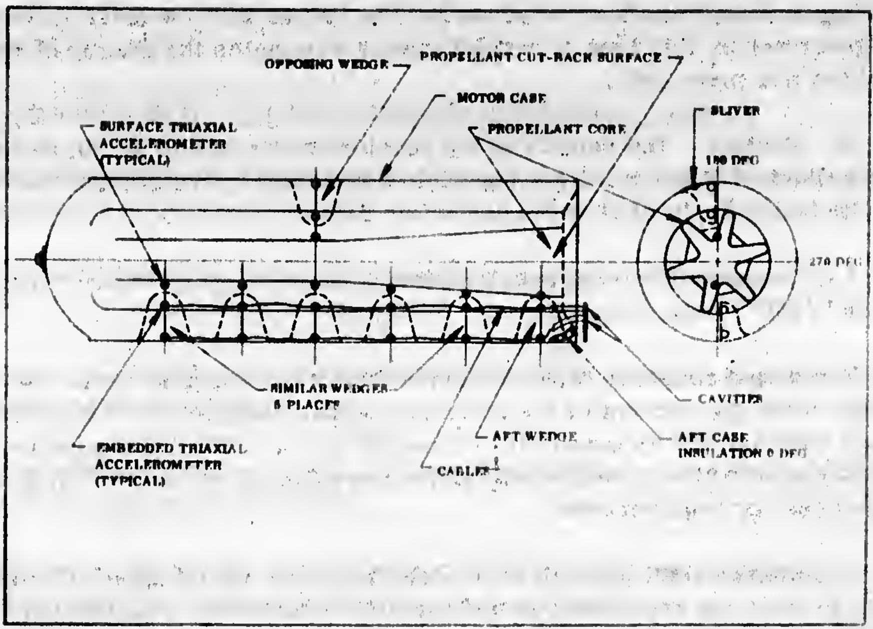

OPPOSING WEDGE
PROPELLANT CUT-BACK SURFACE
MOTOR CASE
SLIVER
SURFACE TRIAXIAL ACCELEROMETER (TYPICAL)
PROPELLANT CORE
180 DEG
270 DEG
SIMILAR WEDGES 6 PLACES
CAVITIES
AFT WEDGE
AFT CASE INSULATION 0 DEG
EMBEDDED TRIAXIAL ACCELEROMETER (TYPICAL)
CABLES

Figure 1. Internal Instrumentation Design Concept

Transducer Selection

After the number and location of the accelerometers were determined, an accelerometer had to be selected that would meet the specifications for dynamic response, temperature, density, and safety. The general specifications for the accelerometers were:

1. Performance - The accelerometers had to contain three sensing elements in a single mounting case, with three mutually perpendicular axes of sensitivity. The sensing elements had to be capable of measuring minimum and maximum sinusoidal accelerations of 0.1 and 50 g peaks, respectively, applied in the direction of the particular axis or axes of sensitivity. The linearity of the response of the sensing elements had to be within ± 1 percent at any frequency between 2 and 2,000 cycles per second. The accuracy of the sensing elements at any frequency between 2 and 2,000 cps within the specified acceleration range, had to be within ± 5 percent. The phase shift between the applied stimulus and the electrical output of the sensing elements could not exceed 1 deg at any frequency or acceleration level within the specified ranges. The distortion of the output wave-form of the sensing elements could not exceed 5 percent of the fundamental signal of an applied stimulus. Cable connectors for the three axes of sensitivity for each accelerometer had to be located on a common side.

2. Safety - Piezoelectric self-generating accelerometers, suitable for embedding in live propellant, were used. The nature and low level of the electrical signal generated by this type of accelerometer eliminates the danger of inadvertently igniting the live propellant.

3. Density - The density of the accelerometer had to be the same as the live propellant, 1.7 gm ± 0.1 gm per cubic centimeter. The accelerometer center-of-gravity had to be located at the geometric center.

4. Temperature - An accelerometer operating temperature range between -20°F and +200°F was required.

The output response of the sensing elements at temperatures other than 70°F had to be within ± 5 percent of the reference output response at 70 degrees. (The response output of the elements was measured with a 2,000 ± 200 picofarad cable). The accelerometer had to withstand temperature cycles up to 300°F without detrimental effects or impairments.

A Gulton triaxial piezoelectric accelerometer, part number TA 320106, was selected to meet the requirements and specifications within a designated time limit. Laboratory tests indicated that these accelerometers could be bonded in the propellant by priming the aluminum outer jacket of the accelerometer with Epon 812 (Epoxy, bond conditioner).

Transducer Cable Selection

An interconnecting electrical cable, compatible with the piezoelectric accelerometer, was selected. Cable requirements were:

1. The cable outer covering or insulation had to be fabricated from a material that could be readily bonded to the live propellant.

2. The overall diameter of the cable had to be kept to a minimum to prevent overstressing the surrounding propellant.

3. The electrical capacitance of the cable could not exceed 40 picofarads per foot of cable.

4. The noise level could not exceed 1 mv peak-to-peak. The noise level was measured by vibrating a 10 ft section, with an initial 3 1/2 in. sag at the center, at 25 cps with a 1 in. double amplitude. During these tests, one end of the cable was connected to the exciter and the other end terminated at a 100 mego impedance source.

After electrically testing cables from several manufacturers, polyvinyl chloride, silicon rubber, and wrapped Teflon jacket materials were selected. An extensive laboratory testing program was conducted to aid in selecting the more suitable cable covering for bonding to live propellant, other factors being equal. A Gulton cable with a silicon rubber jacket, primed with UF-3170 (Silicon Primer), proved to be most compatible.

Accelerometer and Cable Assembly Verification Tests

The following problems associated with the use of the accelerometer and cable as an assembly had to be resolved:

1. What effect did the dielectric properties of live propellant have upon the characteristics of the accelerometer and cable ?

2. What type of cable-to-accelerometer connection configuration was best (connector or integral connection) ?

3. What effect would propellant shrinkage have on the accelerometer and cable assemblies during the cure cycle ?

4. What would be the dynamic transmissibility of the accelerometer and cable assemblies when they were embedded in live propellant ?

Verification Tests

In order to resolve these problems, accelerometer and cable assemblies were embedded in 10 in. propellant cubes for dynamic testing. For one type of specimen, the cable assemblies were routed directly out of the propellant. In a second type of specimen, the cables were coiled prior to routing out of the propellant to determine the effects on accelerometer output when loads were introduced into the cables. The test specimens were bonded to an aluminum plate that also served to attach the fixture to the exciter table. Each of the specimens underwent resonance search tests from 10 to 500 cps at a 1.0 g acceleration level. The samples were also vibrated at 500 cps and 3.0 g for 20 minutes. Side loads were applied to the cable assemblies to evaluate methods of connecting the cable assemblies to the accelerometers.

The maximum frequency at which meaningful data were obtained was 500 cps, due to the frequency limitations of the exciter table. No changes occurred in accelerometer response when side loads were applied to the cable assemblies for either type of specimen. Direct routing of the cables from the specimen was selected as the simpler method of the two. After the verification tests were completed, the specimens were sectioned and inspected for bond separations. The bonds were good in all cases. The dielectric properties of the propellant had no effect on the accelerometer and cable assemblies. Thermal contraction of the propellant during cooldown from the curing temperature did not affect the bond joints or performance of the accelerometer and cable assemblies.

Safety Tests

After final selection of accelerometer and cable assemblies, safety tests were conducted to make sure that the live propellant could not be ignited by connection of the embedded accelerometer cable leads to electrical power supplies. Two types of specimens were prepared for these tests. The first type consisted of small samples of live propellant in which shorted accelerometer cables were embedded. In some of the specimens, the inner conductors were shorted; in others, the shield conductors were shorted, while still others had the inner conductor shorted to the shield conductor.

The second type of specimen was a small sample of live propellant with accelerometer and cable assemblies embedded. After the propellant had cured, the specimens were brought to an ambient temperature of about 80°F prior to testing.

The tests consisted of connecting the cables directly to 115 vac, 220 vac, and 440 vac power sources. The propellant did not ignite, indicating that no hazardous condition exists from the electrical power sources available in the vibration facility.

Embedment Technique

In developing a method of installing a line of three accelerometers at six cross sections along the motor, these criteria were observed:

1. The accelerometer installation and cable routing could not change the dynamic characteristics of the propellant.

2. The accelerometer axes of sensitivity must be oriented within ± 1/4 in. and ± 2-1/2 deg of each other and must be supported by propellant only.

3. The propellant must be cut back without cutting the accelerometer cables.

4. As a secondary objective, the motor would be static test fired after completing the dynamic tests.

Two of the three accelerometers and cable assemblies in each cross section were precast in propellant wedges, which were bonded into the case at each of the linear motor stations prior to casting the propellant. The position and orientation of the accelerometers in the propellant wedges were controlled during the wedge fabrication and were in the same relative position for each wedge (Figure 1).

The wedges were installed so that a common edge of each accelerometer was located on the flat surface of the wedge. The wedges are located linearly in the case by measuring from a common reference point on the motor case. The flat surfaces of the wedges are angularly aligned in the case prior to bonding them in place.

The third accelerometer and cable assemblies required at each of the linear motor stations were installed on the propellant surface prior to attaching the aft closure. The accelerometer orientation was controlled by the mounting blocks used to install them. The mounting blocks, fabricated of balsa wood, were used to orient the accelerometers. The accelerometer and cable assemblies were easily removed after the dynamic tests were completed by shearing the mounting blocks close to the propellant surface (Figure 1).

The cable assemblies from the wedge located at the midpoint of the motor, 180 deg from the continuous row of wedges along the 0 deg target side of the case, were routed circumferentially around the inside of the case. These cables intersected with the cables from the lower accelerometers, which were embedded in the continuous row of wedges. From this point, the cables were bundled together and were routed to the aft end of the motor (Figure 1).

The cable assemblies from the upper accelerometers, which were embedded approximatly 11.5 in. inboard from the case, were routed parallel to the motor longitudinal centerline. The cables were spanned, telegraph style, between wedges and were bonded to the flat face of each wedge (Figure 1).

The cable assemblies from the lower accelerometers, which were embedded approximately 3 in. inboard from the case, were routed along the lined case to the aft end and were bonded to the case liner. All of the cables were tightly bundled before being bonded in position.

By using the vacuum casting process, no voids occurred around the cable bundles during the propellant casting and cure processes. By routing the cables in the manner described, the feasibility of static firing the motor was enhanced, because the flame front would reach the cables in each cable bundle at the same time, and no radial path was provided for a burnthrough or increased burning area (Figure 1).

The propellant is machined to a specified configuration to provide a plenum chamber (Figure 1). This is called a cutback operation. The propellant cutback operation was performed successfully by providing cavities into which the cable assemblies could be placed during this operation. After the cutback operation was completed, the cable assemblies were routed out the aft end of the motor and the cavities around the cable assemblies were filled with propellant (Figure 1).

Bond Joint Tests

Laboratory tests were conducted to evaluate the bond joint between cured propellant wedges and uncured propellant. By priming the cured propellant with Epon 812 prior to the casting operation, the bond joint between cured and uncured propellant was as strong or stronger than the parent propellant. Detailed studies also were conducted to make sure that the properties of the propellant did not change when exposed to three temperature-curing cycles at 135°F of 96 hr duration each. These temperature cycles were required to cure the propellant wedges, the case propellant, and the propellant used to fill the accelerometer cable cavities. Laboratory tests proved that the propellant could withstand these temperature cycles with no physical property changes.

Laboratory tests were also conducted to determine a reliable bond joint between the propellant wedges and the case liner. A UF-2123, ambient-cured liner, was developed and evaluated for bonding cured TP-H1011 propellant on UF-2121 liner. The pot life of UF-2123 liner after deaeration is 4 hr at 75° ± 5°F. Cure times are 96 hr at 75°F, 9 hr at 135°F, and 5 hr at 170°F.

Testing of UF-2123 liner was programed to include all phases of bonding cured propellant wedges in lined motor cases, including preheat prior to casting, vacuum casting conditions, and extended cure at 135°F. Tensile, shear, and fatigue tests were conducted on sample bond joints, utilizing deaerated UF-2123 liner. In all cases, the bond joint had greater strength than the propellant. Other test samples were prepared, using the same systems curing and testing conditions, except the UF-2123 liner was not deaerated prior to application. When these samples were tested, they gave lower values and swelled during the vacuum phase of testing. In all cases where the UF-2123 liner was not deaerated and tested under vacuum, failure occurred in the UF-2123 liner.

Additional test specimens were fabricated to simulate the bond joint between the propellant wedges and case liner. Various loads were applied to these test specimens when exposed to motor processing environment, and in all cases the bond joint was good.

The test data verified that UF-2123 liner was an adequate adhesive for bonding cured TP-H1011 propellant to cured UF-2121 liner. These bond joints were satisfactory for either bayonet-type casting or vacuum casting; however, when they are used with vacuum casting, the UF-2123 liner must be deaerated prior to using.

Special Equipment Requirements

Propellant Wedge Molds

The propellant wedges were designed to support two accelerometer and cable assemblies in a specific orientation and location during casting operations. The final wedges weighed approximately 45 lb and were approximately 14 by 11 by 15 inches.

Fiberglass molds were designed and built to fabricate these propellant wedges. Three different mold configurations were built to meet all the requirements for instrumenting a motor. The first configuration was for the case wedges, the second configuration was for the aft case wedge, and the third configuration was for the opposing case wedge (Figure 2 and reference Figure 1).

The first configuration excluded the case slivers at that location. The slivers were deleted because the close installation tolerance between wedges could be controlled more easily without them. The deletion of this one sliver will not appreciably affect the dynamic response of the propellant or motor (Figure 3 and reference Figure 1).

The second configuration is designed to fit over the aft case insulation and contains the cable cavity connections (Figure 4 and reference Figure 1).

The third configuration fits over the sliver on the opposite side of the motor. Because only one wedge was installed on this side of the motor, the sliver did not interfere with installation tolerances (Figure 1 and 2).

The exact position and orientation of the accelerometers in the propellant wedges were accomplished with holding fixtures specifically located on the wedge mold. The accelerometer and cable assemblies were held in position by these fixtures while the propellant was being cast around them (Figure 5).

Figure 2. Propellant Wedges Installed in Lined Motor Case

Figure 3. Typical Propellant Wedge Installation

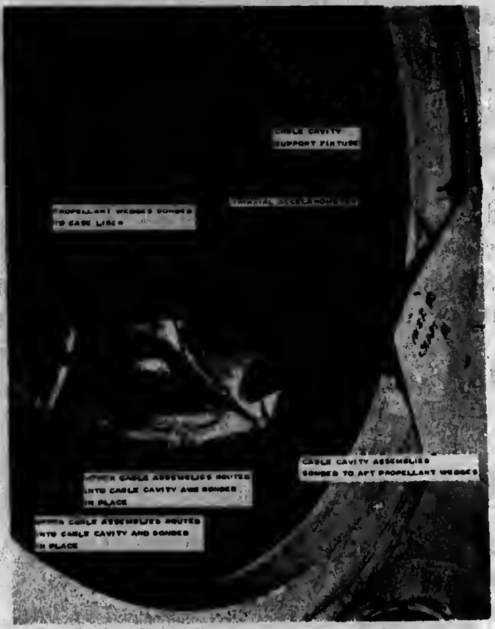

CABLE CAVITY SUPPORT FIXTURE
TRIAXIAL ACCELEROMETER
PROPELLANT WEDGES BONDED TO CASE LINER
OTHER CABLE ASSEMBLIES ROUTED INTO CABLE CAVITY AND BONDED IN PLACE
CABLE CAVITY ASSEMBLIES BONDED TO AFT PROPELLANT WEDGES
OTHER CABLE ASSEMBLIES ROUTED INTO CABLE CAVITY AND BONDED IN PLACE

Figure 4. Aft Propellant Wedge Installation

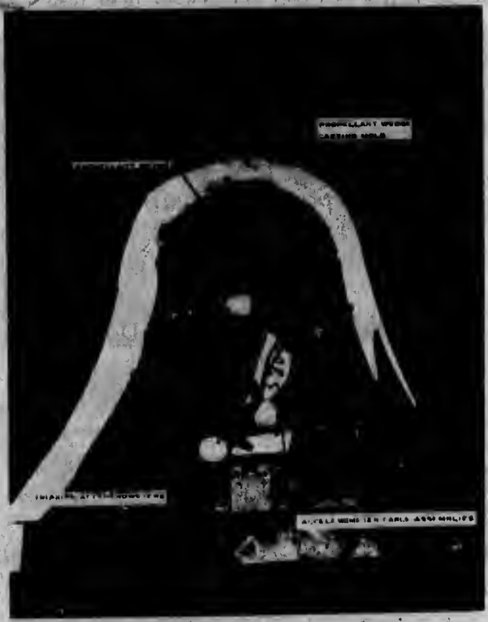


Figure 5. Propellant Wedge and Casting Mold

The major problem associated with the wedge mold design was developing a technique to remove the propellant wedges from the molds after they had cured. Laboratory tests indicated that a conventional rubber-type parting compound would contaminate the propellant wedge and would prevent a good bond joint between the cured and uncured propellant. Attempts to remove the wedges from the mold with air ports in the molds and suction cup pulling devices proved to be unsatisfactory. The purpose of the air ports was to provide a means of breaking the bond between the wedge and the mold. This technique proved unsatisfactory because the air would escape along a very narrow path and would not cause separation of the wedge and mold over a sufficiently large area. The suction cups were satisfactory, but no separation occurred between the wedge and mold. This removal problem was overcome by lining the molds with Teflon tape prior to casting the wedges. When used in conjunction with the air ports, the Teflon tape acted as a good parting material, and the wedges could easily be removed from the molds. No contamination problems occurred when Teflon tape was used.

Cavity Assemblies and Support Brackets

In order to perform the cutback operation, a cavity was provided in which the accelerometer cable assemblies could be safely placed. After performing calculations and experimental tests on different methods of coiling the accelerometer cables, a cavity approximately 15 in. long by 2-1/2 in. in diameter was found to be satisfactory. These cavities were formed by casting the propellant around cavity assemblies (aluminum tubes). After the wedges had been installed in the case, the accelerometer cable assemblies were routed from the last wedge into the cavity assemblies, and the joint between the cavity assemblies and the wedge were sealed (Figures 6 and 7).

The cavity assemblies were wrapped with Teflon tape to simplify removal after the propellant had cured. A support bracket was designed and fabricated to hold the cavity assemblies in position during propellant casting. This support bracket restrained any axial loading of the cavity fixtures due to buoyancy when the fixtures were submerged in uncured propellant (Figure 4).

Manufacturing and Processing Details

Preparation of Accelerometers and Cable Assemblies

The accelerometers were all calibrated electrically, and the calibration results for each axis of sensitivity were recorded. Prior to embedding the accelerometers in propellant, they were primed with Epon 812 Liquid Epoxy Resin to insure a good bond with the propellant. Cable assemblies for each accelerometer axis of sensitivity were fabricated and identified with a coding number. The coding numbers were required to identify the embedded accelerometers and corresponding axis of sensitivity after completing the casting operations. The cable assemblies were

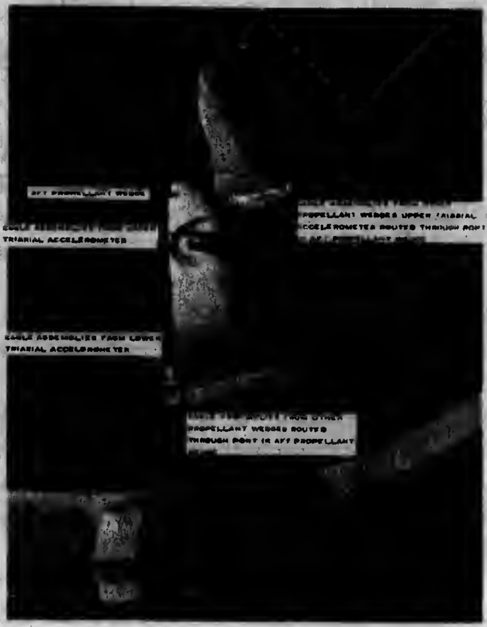

AFT PROPELLANT WEDGE
TRIAXIAL ACCELEROMETER
PROPELLANT WEDGES UPPER TRIAXIAL ACCELEROMETER ROUTED THROUGH PORT
CABLE ASSEMBLIES FROM LOWER TRIAXIAL ACCELEROMETER
FROM OTHER PROPELLANT WEDGES ROUTED THROUGH PORT IN AFT PROPELLANT

Figure 6. Cable Routing at Aft Propellant Wedge

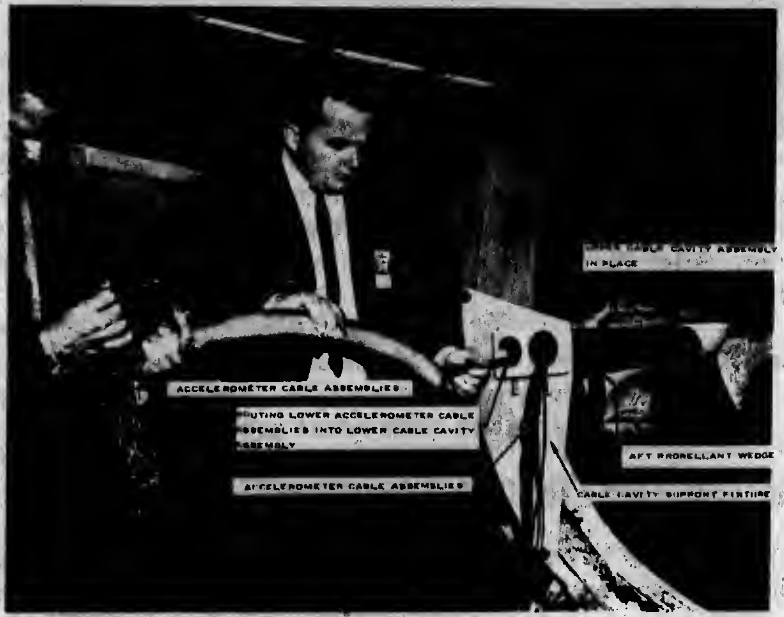

CAVITY ASSEMBLY IN PLACE
ACCELEROMETER CABLE ASSEMBLIES
UTING LOWER ACCELEROMETER CABLE SSEMBLIES INTO LOWER CABLE CAVITY SSEMBLY
AFT PROPELLANT WEDGE
ACCELEROMETER CABLE ASSEMBLIES
CABLE CAVITY SUPPORT FIXTURE

Figure 7. Installation of Accelerometer Cables in Cavity Assemblies

connected and safety wired to the accelerometer. The cable assemblies were primed with UF-3170 sealant to insure a good bond when the assemblies were embedded in the propellant.

Installation and Electrical Tests of Accelerometers and Cable Assemblies

The accelerometer and cable assemblies were installed in the wedge and were checked electrically prior to casting the wedges. These electrical tests insured that the accelerometers or cables were not shorted or broken prior to embedding them in the propellant wedge. If any defect had existed, repairs or replacements at this stage of the process easily could have been made.

The accelerometer cable used was unique in that it had nearly zero series resistance and a distributed capacitance of 33.6 picofarads per foot of cable. With this capacitance characteristic, the amount of capacitance a cable of a given length will have is known. By adding the capacitance of the accelerometer, the total capacitance of any accelerometer-cable combination is known. One of the electrical tests utilized was to measure the capacitance of the accelerometer-cable combinations. If the cable or accelerometer circuit is broken, it is easily detected and the specific location of the break is known within inches.

An additional electrical check was required because the cable or accelerometers could have had a high resistance short that was not detected by the capacitance test. By putting a 50 vdc input into the accelerometer-cable combinations and measuring the resistance between the inner conductor and outer shield, any existing shorts were detected.

Propellant Wedge Processing

After completing the electrical checks and lining the wedge molds with Teflon tape, the wedges were cast with propellant. Two small individual propellant batches were used to cast the propellant wedges. The propellant wedges were bayonet cast instead of vacuum cast to permit manual control of the cable routing and the flat surface of the wedge. After the wedges were cast, they underwent a cure cycle of 135°F for 96 hours.

After curing, the propellant wedges with the embedded accelerometer and cable assemblies were removed from the casting molds. The propellant wedges were individually X-rayed to verify that no voids or air pockets existed.

Installation of Propellant Wedges and Cable Assemblies

The propellant wedges initially were installed dry, positioned, and aligned. With the wedges in the proper position, the case liner was masked off so that the wedges could be quickly installed in the correct location after the bonding compound was applied. Because of the confined work area, only one wedge was handled at a time and always the extreme aft wedge. Therefore, after the locations were masked off, all the wedges were removed prior to permanently installing them with bonding compound (Figures 8 and 9).

As each wedge was installed with bonding compound, its alignment was checked visually. After the opposing wedge on the 180 deg axis of the case was installed and aligned, the UF-2123 bonding compound was allowed to cure for 96 hr at ambient temperature (75°F). After the bonding compound on this wedge had cured, the motor was rotated 180 deg and the remaining wedges were installed in the same manner. The cable assemblies from the opposing wedge then were routed circumferentially around to the other wedges and were bonded to the case liner with UF-2123 liner. The motor was rotated as required during this operation. The cable assemblies from this wedge and the bottom cable assemblies from all the other wedges were routed along the case and into the lower cavity assembly (Figure 1).

An electrical check was made before bonding the cables to the case liner to insure that the cables had not been damaged during the processing, X-ray, and installation operations. Any damaged cable assemblies were repaired prior to bonding them in place.

The lower cable assemblies were bonded in place, and the point between the lower cable cavity assembly and the aft wedge was sealed with UF-2123 liner (Figure 4).

After the lower cable assemblies were bonded in place, the motor was rotated 90 degrees. The upper cable assemblies were bundled and routed along the wedge and into the upper cavity assembly. Rotating the case made it easier to bond the cable assemblies to the propellant wedges (Figure 10).

An electrical check was again conducted before bonding these cables to the propellant wedges.

The upper cable assemblies were then bonded in place, and the joint between the upper cable cavity assembly and the aft wedge was sealed with UF-2123 liner (Figure 4).

All of the installed propellant wedges were primed with Epon 812 to insure a good bond between the cured propellant wedges and the remainder of the motor propellant.

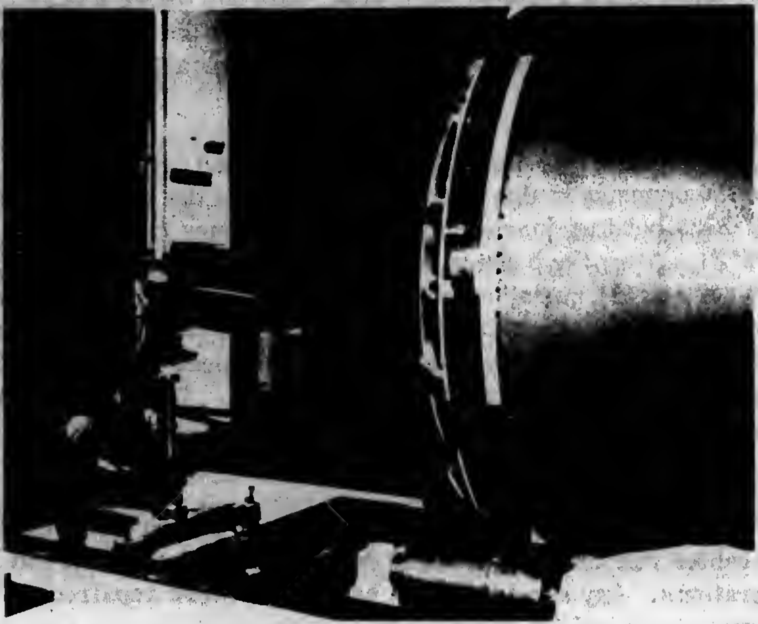


Figure 8. Propellant Wedge Alignment Operation

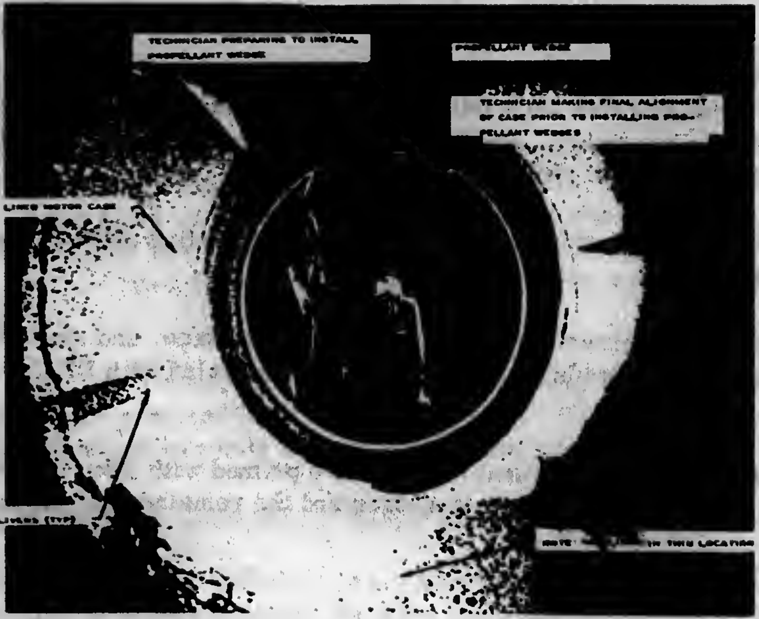

TECHNICIAN PREPARING TO INSTALL PROPELLANT WEDGE
PROPELLANT WEDGE
TECHNICIAN MAKING FINAL ALIGNMENT OF CASE PRIOR TO INSTALLING PROPELLANT WEDGES
LINED MOTOR CASE
SLIVERS (TYP)
NOTE: [illegible] IN THIS LOCATION

Figure 9. Propellant Wedge Installation

N HORIZONTAL POSITION PRIOR TO
R ACCELEROMETER CABLES TO THEM
LEROMETER CABLE

Figure 10. Positioning Upper Accelerometer Cables Prior to Bonding

Wedge Installation Tests

Before casting a full scale motor with internal instrumentation, verification tests were performed utilizing an aft subassembly. The bond joint was checked by applying weights of 27, 18, and 10 lb to the propellant wedges installed in the aft subassembly. The 27 lb weight, the most severe condition, reduced the safety factor from 8.32 to 2.83 (Figure 11).

The weighted wedges were exposed to temperature and vacuum conditions similar to those encountered during full scale motor casting. No failures occurred during this temperature-vacuum cycle; therefore, the weights were removed and the subassembly was cast with propellant as a final confirmation of the design concept. The X-ray and visual inspections of this specimen after the propellant had cured proved conclusively that the concept was technically sound.

Casting and Curing Motor

After successfully completing the wedge installation tests in the aft subassembly, the full scale motor was prepared for processing. The wedges were installed in the lined case and the motor was prepared for casting according to normal operating procedures. The casting core, the vacuum casting bell, and other miscellaneous equipment were installed (Figure 12).

The motor casting and curing operations were accomplished with no problems or unusual occurrences. After curing, the cable cavity assemblies were removed, and the accelerometer cables were coiled and placed in the cavities below the cutback depth. The motor was cut back in accordance with normal processing procedures.

Figure 11. Weight Installation on Propellant Wedge

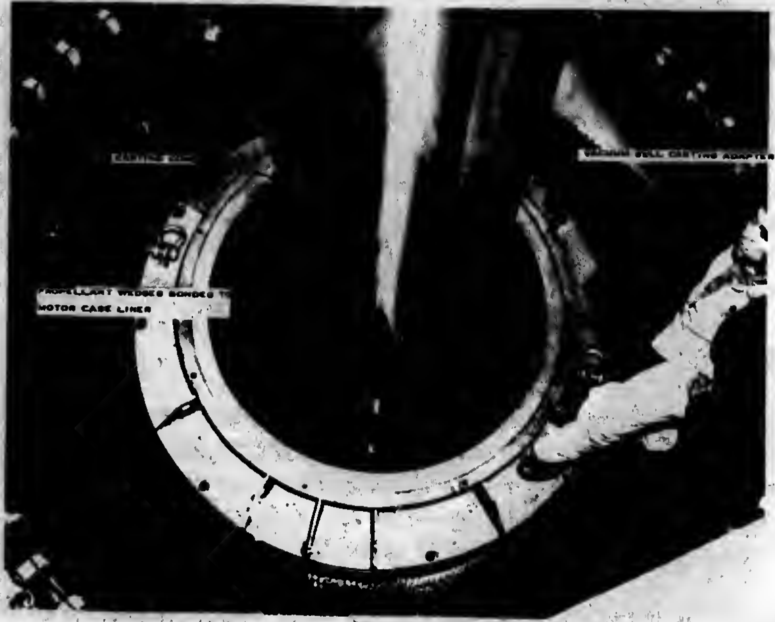

CASTING CORE
VACUUM BELL CASTING ADAPTER
PROPELLANT WEDGES BONDED TO MOTOR CASE LINER

Figure 12. Casting Core Installation

The accelerometer cable assemblies then were uncoiled and routed directly out of the cavities. The cavities were primed with Epon 812 Liquid Epoxy Resin prior to filling them with TP-H1064 propellant. This propellant is identical to TP-H1011 except that it has spherical aluminum particles instead of coarse ground aluminum particles. This propellant is less viscous in the uncured state than TP-H1011 and does not develop as many voids when filling small cavities. After filling these cavities, the motor was recycled at 135°F for 96 hr to cure the propellant.

Inspection and Final Processing

The motor was given a complete X-ray and ultrasonic inspection prior to installing the aft closure and nozzle assemblies. No voids or cracks were revealed in the propellant or around the wedges or cable assemblies.

Final electrical checks showed that all the accelerometer and cable assemblies were completely functional.

Example of Results

Longitudinal Mode of Vibration

The first longitudinal resonance of the solid propellant rocket motors tested occurred at 37 cps. The corresponding mode shape and phase relationships shown in Figure 13 indicate that the predominant response of the case and propellant is that of a fixed-free rod with a slightly coupled propellant thickness shear mode.

The second longitudinal resonance of the solid propellant rocket motors tested occurred at 70 cps. The corresponding mode shape and phase relationships shown in Figure 14 indicate that the predominant response of the case and propellant is that of a free-free propellant thickness shear mode with a slightly coupled case rod mode.

Transverse Mode of Vibration

A typical example of the mode shapes assumed by a rocket motor case and different levels of the propellant while vibrating the motor at its simply supported beam bending frequency, which is an antiresonant condition is shown in Figure 15.

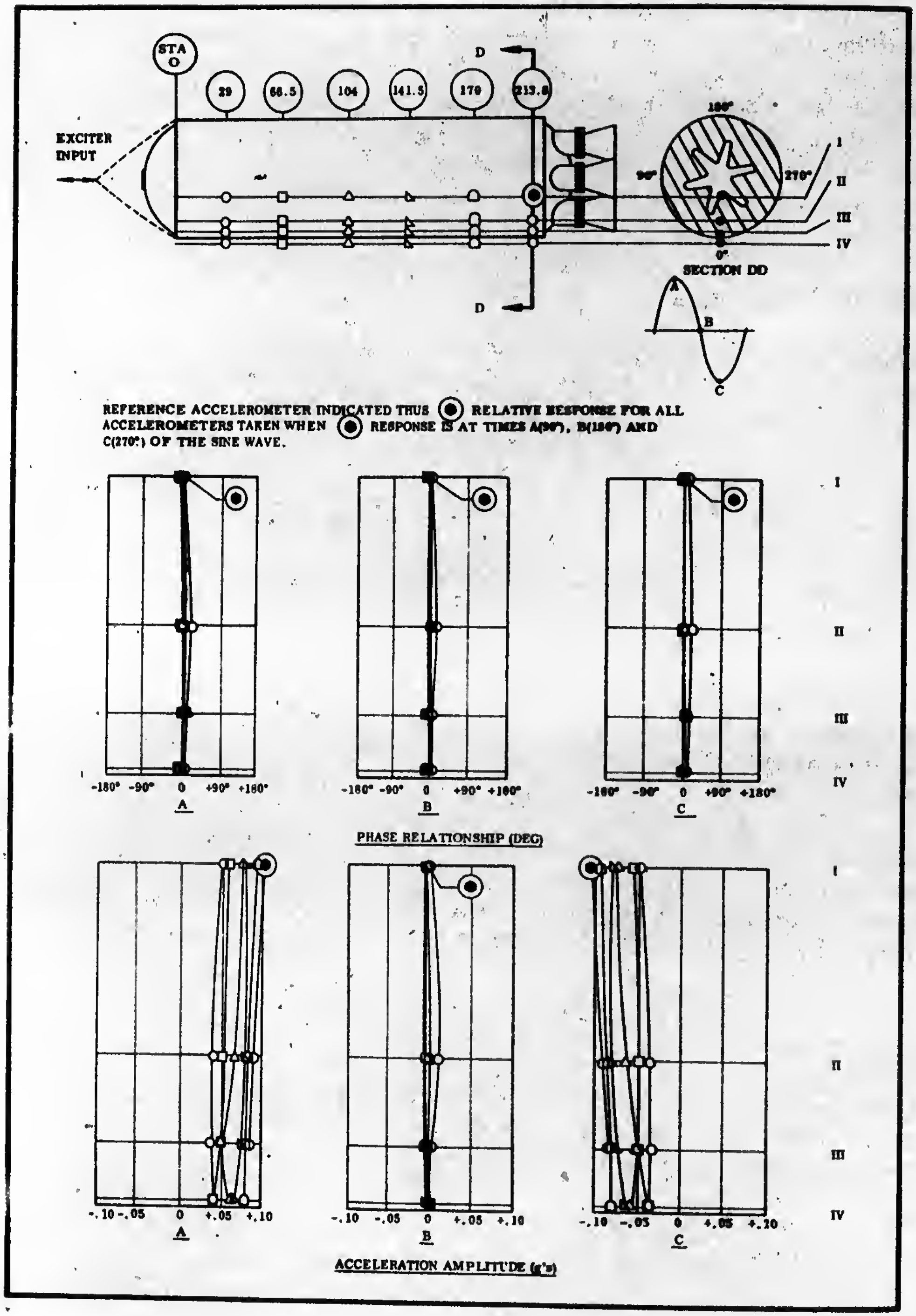


Figure 13. Longitudinal Response of Motor Case and Propellant at 37 cps

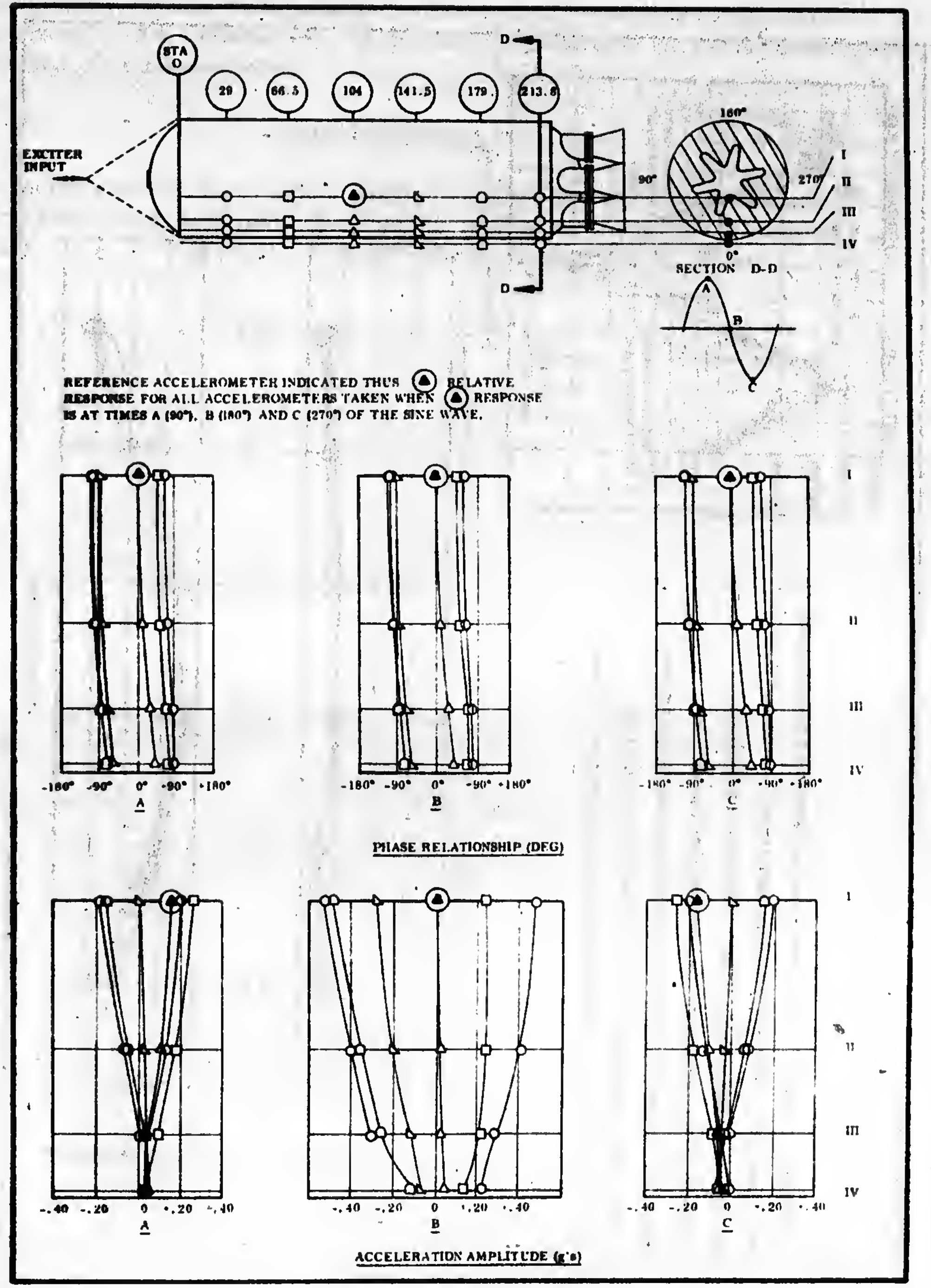
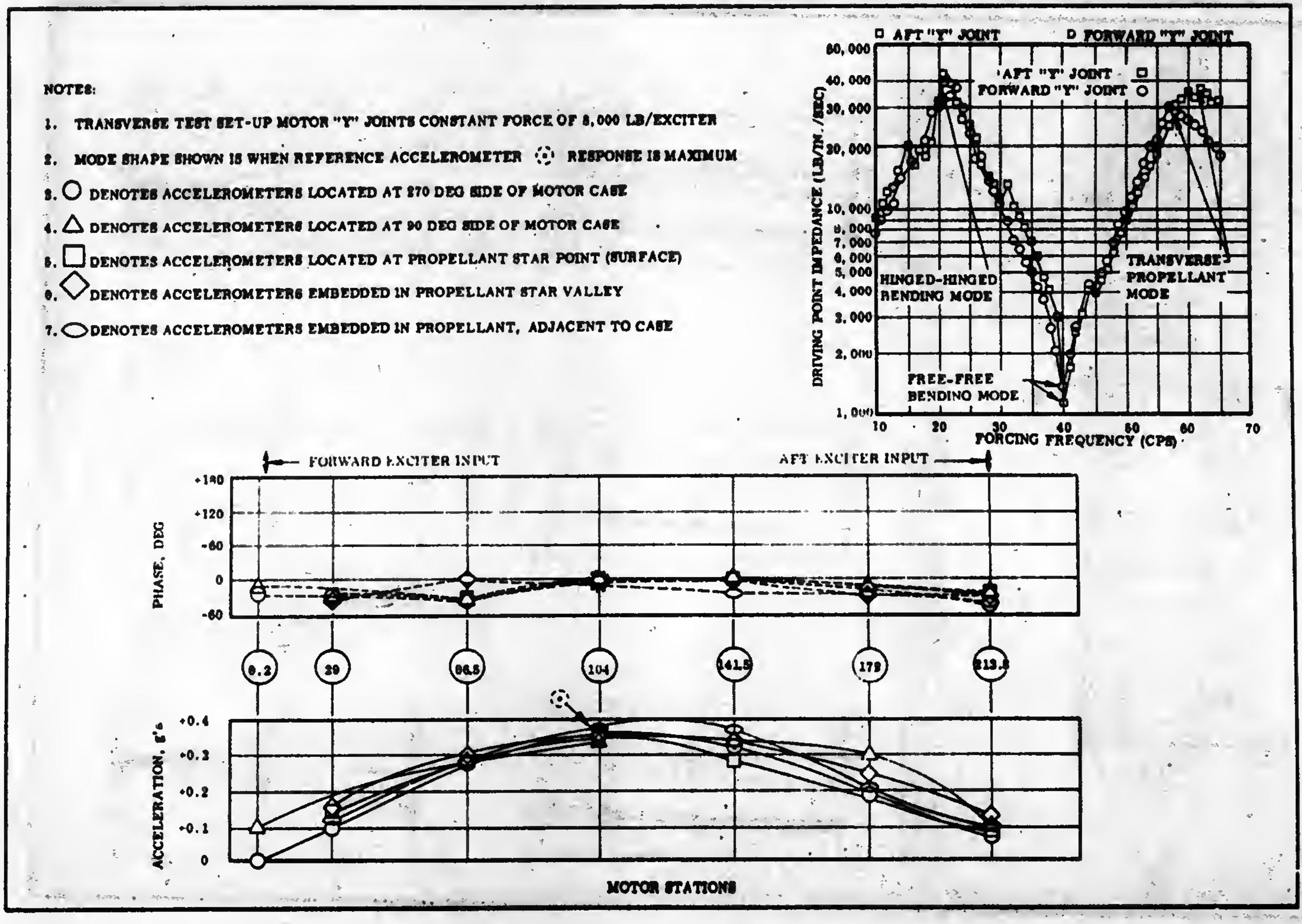


Figure 14. Longitudinal Response of Motor Case and Propellant at 70 cps


NOTES:

1. TRANSVERSE TEST SET-UP MOTOR "Y" JOINTS CONSTANT FORCE OF 8,000 LB/EXCITER
2. MODE SHAPE SHOWN IS WHEN REFERENCE ACCELEROMETER ◌ RESPONSE IS MAXIMUM
3. ○ DENOTES ACCELEROMETERS LOCATED AT 270 DEG SIDE OF MOTOR CASE
4. △ DENOTES ACCELEROMETERS LOCATED AT 90 DEG SIDE OF MOTOR CASE
5. □ DENOTES ACCELEROMETERS LOCATED AT PROPELLANT STAR POINT (SURFACE)
6. ◇ DENOTES ACCELEROMETERS EMBEDDED IN PROPELLANT STAR VALLEY
7. ⬭ DENOTES ACCELEROMETERS EMBEDDED IN PROPELLANT, ADJACENT TO CASE

Figure 15. Transverse Response of Motor Case and Propellant at 21 cps

The technical results of the vibration tests are not presented in this paper. The typical examples are included only as an illustration of what may be accomplished with this method of instrumentation.

Acknowledgements

The authors would like to acknowledge the valuable contributions and efforts of these personnel in the successful development of this instrumentation technique: B. D. Bryner, J. O. Campbell, L. Gammel, E. E. Meeker, R. E. Meyer, R. J. Porter, and T. Sedgwick.

ENERGY STORAGE AND DELIVERY SYSTEMS, AN URGENT REQUIREMENT FOR AEROSPACE CONTINGENCIES.

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A few months ago, a major disaster at sea startled the world. The U.S.S.'Thresher' was lost with all hands, in waters which are among the best charted and best known in the whole world and a few miles only from the main base of operations. In the absence of really adequate data, it has been, so far clearly impossible to reconstruct a proper explanation of the tragedy except to learn that, at a comparatively trivial depth compared with that of the world's oceans means of reconnaissance for the wreck and for rescue under such circumstances, have been either extremely inadequate for the task or totally lacking.

If a survey is made of the increasing toll of flying accidents both in the civil transport and in the military field it must be conceded that there are, yet, many equally inexplicable losses of life and of materiel in regions of the troposphere and lower stratosphere where the problems of flight ought to be well understood.

The nature of these experiences is such that their study and their resolution are of importance to astronautics which has profited by analogies in many fields. Criteria which are used to gauge the performance of sea going vessels and, more recently, aircraft have been expressed in standards which have been appropriately codified by marine surveyors or by aircraft inspectors, and the concepts of sea-worthiness and airworthiness are well understood. As yet, however, in spite of several orbital and sub-orbital flights successfully achieved in this country and in the Soviet Union, and clearly a successful medical and training programme for personnel, there is, so far, little visible evidence of a realistic or concerted drive to promote really systematic study of the many criteria which must be implied in true 'spaceworthiness of manned space vehicles.

Just as the analysis of data, on an international scale of co-operation has promoted transport successes in other spheres it should now be possible to infer that favourable results for astronautics will be achieved by properly concerted action on an international scale. A new discipline of Astrography may result which would be a counterpart of Oceanography. It would be a specially analytic field in which the space sciences would be considered from the point of view of application rather than theory.

Numerous contributions have already been made some of the most significant and relevant having been studies founded on the extension of radiobiology and of radiation damage already available into the space environment, suitably amplified by newer findings. By far the most basic problem however hinges on propulsion.

Chemically fuelled rockets which have been dominating aerospace research up to the present have had the great advantage of being largely independent of their environment but this attribute which has been so valuable has tended to obscure their most serious limitations when the choice of optimum prime movers for serious and protracted voyages into deep space are under consideration. Their bulk and short endurance obviously render them obsolete for such problems.

The investigations currently proceeding over a wide front into such questions as the acceleration of electrically charged fluid masses under conditions of ionic flow, or of the transport of plasma by pumping fields point the way towards a realization of the desired techniques which would provide an adequate supply of power located midway between the extremes of the high energy high thrust chemical rockets of short duration, and the low-energy low thrust devices such as ion motors or plasma jets so far developed to cater for long endurance, cruise conditions. There is plainly a most serious 'propulsion gap' which needs to be filled before soundly conceived manned power flight in space can be contemplated.

As is usual in such problems involving transport phenomena we may divide the problem into;

a. The means by which the chosen form of energy source may be made to exert a force on its environment. This is the problem of the working substance.

b. The means by which acceleration of the working substance may be achieved.

c. Associated with the above is the question of energy storage. This may generally be considered in terms of thermodynamic potentials, of dynamic or electrical storage or, ultimately, the role of matter itself as form of energy rather than a vehicle through which it acts.

So far, one of the most impressive and, indeed, promising methods aimed at the resolution of the 'Propulsion Gap' has been the Gas Core Nuclear Rocket and consideration of this is desirable.

Gas Core Nuclear Rockets.

The present position of nuclear rocket development is dominated by the attendent technologies of civil power generation stations and naval reactors from which much useful data has become available. At the same time these also restrict the development of suitable space-borne react-representing as they do a large vested interest in man-power and indeed concepts centred on quite different working conditions.

Typical layouts of gas core reactors are shown in Figs 1,2,[illegible]. It will be noted that, as in a normal bi-fuel rocket, two gaseous components flow into a chamber consisting of a cavity surrounded by a moderating shield. The propellent generally considered fo be most promising is hydrogen and a possible fuel which has received a good deal of study has been UF_6 containing the fissionable isotope of uranium, As envisaged such a device, presupposing that flow and containment could be sustained, would operate at exhaust gas temperatures of 10,000 deg.R to 15,000 deg.R. and specific impulses of up to 3,000 sec. are planned. Performance and endurance calculations by Ragsdale and others have shown such a figure for specific impulse would correspond with a thrust weight ratio which would enable a vehicle designed for a round trip to a planet such as Venus with a total vehicle weight of about half of that obtaining in the case of a solid cored reactor rated for a similar mission. (1,2,3,4,5).

Because of the extremely high pressures which would obviously be necessary in a reaction of this kind, there is a body of opinion which considers that there must be an induced differential between the flow rates of the two gaseous components as they pass through the cavity which has been formally expressed in terms of 'residence times' This implies that there is a Separation Factor which is the ratio of the hydrogen velocity through the reactor to that of the fissionable fuel.

It has been shown that;

$$\frac{P}{1{,}000} = \frac{W'}{35} \cdot \frac{N_p}{10^{18}} \cdot \frac{T}{10{,}000}$$

S is Separation factor.

N_p is the critical density at/c.c.

T is the reaction temperature deg.R.

W' is the hydrogen to fuel flow-rate ratio.

With a separation factor of 100 there seems to be a theoretically feasible reactor type but the engineering problems are formidable and methods of bringing about partial separation would clearly be extremely complex.

Two methods which have been proposed which might provide a basis for operating have been;

a. Vortex containment systems which may be single or multiplex. (1,2,3,4.)

b. CO-axial flow systems. (2)

In addition, some attempts have been made at JPL to analyse the conditions of operation of a plasma-core type reactor. So far the bulk of the work has been of a theoretical nature but some publications of the Russian Kikoin working in this field might indicate that the theory has been applied to an extent. Calculations by G.Safronov have shown that with a cavity radius of about 1 meter using a D_2O reflector a charge of about 2kg. would correspond to a condition of criticality.

These details of cavity and vortex stabilized cavity reactors are cited in this paper because they seem to provide the first and obvious solution to the vital problem of the propulsion gap and there can be little doubt that, in terms of feasibility, of range and of thrust these prime movers will be attainable after particularly difficult development programmes. However they would present extremely difficult problems in habitability of space-craft and of shielding in view of their high temperatures and radiation rates.

While work in this direction is seen to have promise there are a number of other directions in which enquiry should be directed, particularly those in which there seems to be promise offered by acceration of plasma or ionised gases at relatively low magnetic Reynolds numbers at which high temperature, containment and radiation problems are much less formidable. As an introduction to these it is profitable to study the applications of MHD drive to vortex systems such as those suggested by the work of Rozenzweig and his collaborators, noting, however, that their goal has been a different one.(3 There is also the work of Gross and Kessey, who have examined the theory of an MHD plasma centrifuge.(4). In this the plasma is contained between two co-axial cylinders. Fig.( ).

MHD Equations and Analysis of the Plasma Vortex.

Continuity $\nabla \vec{V} = 0$ 2.

Momentum $\rho(\text{[illegible]}\vec{V}/dt) + \nabla p = \vec{J} \times \vec{B}$ 3.

Faraday's Law $\nabla E = -(d\vec{B}/dt)$ 4.

Ampere's Law $\nabla \times B = -\mu \vec{J}$ 5.

Ohm's Law $\vec{J} = \sigma(\vec{E} + \vec{V} \times \vec{B})$ 6.

Divergence Law $\nabla \vec{B} = 0.$ 7.

V = fluid velocity
B = magnetic field vector
E = electric field vector
ρ = bulk fluid density
σ = conductivity
μ = magnetic permeability.

From the above basic equations, equations for mass flow in a long cylinder may be derived, using a single fluid density;

$$E_r = (I_r / 4\pi h_0 \sigma r) - V_0 B_0$$

$$-\frac{\rho_0}{r}\left(\frac{\partial V_\theta}{\partial r} + \frac{V_\theta}{r}\right) + \eta \frac{\partial}{\partial r}\left[\frac{1}{r}\frac{\partial}{\partial r}(r \cdot V_\theta)\right] - \frac{B_0 I_r}{4\pi h_0 r} = 0$$

$$\frac{1}{\mu\sigma}\frac{\partial}{\partial r}\left[\frac{1}{r}\frac{\partial}{\partial r}(rB_\theta)\right] - \frac{G_0}{\rho r}\left(\frac{\partial B_\theta}{\partial r} - \frac{B_\theta}{r}\right) = 0$$

$$\frac{\partial p}{\partial z} = -\frac{B_\theta}{\mu}\frac{\partial B_\theta}{\partial z}$$

$$\frac{\partial p}{\partial r} = \rho\left[\left(\frac{G_0 r_0}{\rho}\right)\frac{1}{r^3} + \frac{V_\theta^2}{r}\right] - \frac{B_\theta}{\mu r}\frac{\partial}{\partial r}(rB_\theta)$$

For a complex fluid such as a partially ionized, incompressible plasma (electrons, light and heavy ions light and heavy neutral particles.) ;

$$\rho = \sum_{i=1}^{5} n_i \cdot m_i = \text{const.}$$

n_i = number density of species 1

m_i = mass of particle of the [illegible] species

A solution is obtainable from detailed consideration of the above relationships defining the electric power required to rotate plasma per unit mass, which is;

$$\frac{P_m}{\eta\,\phi_0^2/\rho B_0 r_0^4} = \frac{4\,|R_N|}{1-\xi_1}\,\bar{R}\,.$$

where

$|R_N|$ is radial Reynolds no

P_m total electric power input

η is absolute coefficient of viscosity

ϕ_0 electrical potential difference across electrodes.

ρ density

B_0 is applied axial magnetic field

r_0 is radius of outer electrode

ξ is r/r_0

and
$$\bar{R} = \frac{2(m+1)}{[m+1]\left[\xi_1^2 - 1 + 2\left(\frac{2R_N}{M^2} - a_2\right)\ln\xi_1\right] - 2a_1\left(\xi_1^{m+1} - 1\right)}$$

while

$$m = 1 + R_N$$

Typical results which have been obtained from studies on gaseous core rockets are represented by the following data;

Exhaust gas temp. = 11,000 deg. R.
Average gas press. = 500 atm.
Reactor volume = 10.3 cubic metres
Total power = 1.21×10^6 Kw.
Reactor radius = 1 metre
Radial current = 348 amperes.
Voltage diff. = 50 volts.
Axial magnetic field = 10^4 gauss.
Mass flow = 5.63 lb. sec.
Thrust = 16,900 lbs.

The thrust is still comparatively small and would have to be considerably increased in order to justify the effort that will be needed to realize serviceable equipment working on these principles.

Rosa has pointed out in a recent study of a combination of a cavity reactor with an MHD generator, that the cavity reactor as discussed above, suffers from the rather important disability that a good deal on unfissioned uranium would be dumped in the exhaust and that full efficiency could only be maintained if this could, somehow, be recovered and recycled. This rather difficult task could be obviated, he believes, by the use of the combined cavity reactor and MHD generator. (8).Figs.

As illustrated, the installation also includes an MHD generator of a kind now considered to be practicable after a good many development difficulties have been resolved. It would produce, in conjunction with a cavity reactor about 300 tons of thrust and operate with a specific impulse of about 1,000 seconds. The operating chamber pressure and temperature would be 100 atmospheres and 300 deg.K. using a working fluid consisting of hydrogen containing 1% U_{235} and 1% Potassium vapour. For criticality, it would require a diameter of about 2 metres and would need a D_2O blanket of about 7 tons weight. Gas escape temperature would be about 2,000 deg. K. and pressure would be atmospheric after a 100/1 expansion. The MHD generator working from these exhaust gases would be designed for a gas flow of 1.45×10^5 mols/sec and for a power output of 14,000 megawatts. This is, of course, high compared with existing power generating plant but is not high compared with output achieved with chemical rockets for short periods. It should be quite adequate to put a 100 ton satellite into close orbit and would be invaluable on longer space missions on which there would be a requirement for substantial power to deal with contingencies.

Study of MHD generators of the above type is leading to new approaches to the study of plasma and highly ionized gases both as a source of energy through fusion reactions and as means of storing energy which may be drawn off for propulsion. It is becoming better understood that the properties of plasma in a magnetic field in which it behaves as a diamagnetic medium ($k_m < 1$) depending for its conductivity on the

Energy as it may be used for purposes of propulsion may be stored in the following ways;

1. As chemical energy, fuels; phenomena of combustion dissociation etc.

2. As nuclear energy, fission, fusion or other processes.

3. Momentum as in flywheels.

4. Electromagnetic energy, currents flowing in inductances.

5. Electrical energy in condensers or capacitors.

If W_0 represents energy density, i.e. Joules / cm^3, we have been able to obtain from chemical explosives $W_0 \simeq 10^4$ / cm^3. In order to obtain a similar density in a magnetic storage system, the field strength H would need to be $\simeq 1.6 \times 10^6$ gauss. If one considers the rim of a flywheel made of steel moving at the periphery at sonic speed, the kinetic energy at the rim is 500 Joules / cm^3. There would be an upper limit here imposed by the strength of the material. In condensers, one may have an energy density of about 10^{-2} Joules / cm^3, much below the value appertaining to a flywheel. Much attention has been paid recently to the improvement of electrical engineering materials and with improved oxides of titanium with exceptionally high dielectric constants it is hoped to be able to produce condensers with the capability of storing a charge of over 1 Joule / cm^3. It would appear that the discovery of the optimum conditions of conferring energy of motion to charged gases probably in the comparatively stable configuration of the vortex presents one of the best alternatives needed to resolve the propulsion problem. It would also involve means of converting and diverting the momentum thus stored into linear acceleration. A survey of the principal natural processes of large scale transport phenomena such as movements of air masses or of portions of the solar corona would seem to stress the ultimate feasibility of the concept.

Electomagnetic Plasmoid 'Whirls'

It is well known from the established theory of magnetohydrodynamics that;

$$J = \quad (\vec{E} + \vec{V} \times \vec{B})$$

Where J = Current density

σ = Conductivity of the gas

$\vec{B}$ = Magnetic field vector

V = Velocity.

$\vec{E}$ = Electric field vector.

This is a special form of Ohm's law applicable to plasma transport.

In addition, the Magnetic Reynold's Number R_m is defined by

$$R_m = \mu_0 \sigma v l$$

μ_0 is permeability of free space

σ V are as above ; l is a linear dimension

There are generally two main regions to be considered when the question of moving ionized gases or plasmas are under consideration. They are those in which ; (1) $R_m \gg 1$ and, (2) $R_m < 1$ 1. Fig.

(1) This is the region of thermal or equilibrium ionization occurring at high temperatures and is typical of the conditions obtaining during the investigation of fusion devices.

(2) This is the region of extra-thermal or non-equilibrium ionization and which has long been known in connexion with investigations on low pressure gas discharges.

There would appear to be a region on the general curve (Fig 0 lying below the point of inflexion of the curve which has not received much attention but which has been under consideration in connexion with the setting up of rotating plasmoid systems which have been named 'whirls' by Japolsky who has carried out an extensive investigation on their properties. Their conception and relationship with e/m wave systems rotating in vacua arose from analysis of propagation from a wave guide under particular conditions. They are particular solutions of the Maxwell equations.

The Maxwell equations may be set out as follows;

$$\vec{E} = C \,\mathrm{curl}\, \vec{B} \quad ; \quad \vec{B} = \mathrm{curl}\, \vec{E}$$

$$\mathrm{div}\, \vec{E} = 0 \quad ; \quad \mathrm{div}\, \vec{B} = 0$$

$\vec{E}$ is electric field intensity vector (e.s.u)

$\vec{B}$ is the Magnetic field intensity vector

dot over vectors indicates d/dt in fixed coordinates and c.d/dt in invariant coordinates.

In addition, solutions of Maxwell's equations must always satisfy the 'wave equations'

$$\Box^2 \vec{E} = 0 \quad ; \quad \Box^2 \vec{B} = 0$$

$$\text{where } \Box^2 = \nabla^2 + \frac{\partial}{\partial(ict)^2} \text{ in fixed coordinates}$$

$$= \nabla^2 + \frac{\partial}{\partial(i\tau)^2} \text{ in invariant coordinates}$$

$$\nabla^2 = \text{curl curl} + \text{grad div.}$$

c is the ratio between the absolute electrostatic and absolute electromagnetic units of field intensity and is usually referred to as the velocity of light.

In the theory we are obliged to take account of two kinds of solutions to the Maxwell equations. These are (1) An elementary solution ('simple whirl). (2) Integral solutions ('compound whirls)

For the 'simple whirl'

$$\begin{cases} \vec{E} = A\beta q_0^2 R(e^{[i(\tau + n\theta_1) \mp \chi\xi]}\vec{q}) \\ \vec{B} = A\beta q_0^2 R(e^{[i(\tau + n\theta) \mp \chi\xi]}\vec{n}) \end{cases}$$

A is the amplitude constant

R is a symbol indicating that only the real portion of the complex expression in the brackets need be taken.

is a non-dimensional constant

$$\beta = \left(1 - \frac{V^2}{c^2}\right)^{-1/2}$$

V is the velocity of motion of the 'whirl' with reference to a coordinate system taken as fixed.

$$q_0 = \frac{2\pi}{\lambda_0} = 2\pi\nu = \frac{\Omega}{c}$$

λ is the wave length of the 'whirl.'

n is its frequency

Ω is its angular frequency with respect to the moving coordinates.

where x is the axial coordinate with reference to the moving coordinate system and may be called an invariant axial coordinate x by the relationship;

$$x = x_1 + vt$$

t is time in seconds

τ is the invariant time coordinate $\frac{\Omega_0 t}{\rho}$ $\frac{\tau'}{\beta}$

γ is a non-dimensional constant which characterises the exponential fall in the field intensity in the axial direction r.

$$\gamma = \frac{|K|}{\beta q_0} \equiv \text{exponential decrement fn.}$$

The upper sign (-) relates to $\xi > 0$, i.e. the space in front of the central plane (=0) while the lower sign (+) relates to (= 0), i.e. space behind the axial plane. Also;

$$\theta_1 = \theta - \frac{v}{c}\frac{\xi}{r} = \theta \mp \frac{\beta q_0}{r} x$$

where θ is the angular or azimuthal coordinate, i.e. the invariant angular coordinate.

$\vec{G}$ and $\vec{H}$ are vectors depending on the radial coordinate above.

Of especial significance to the theory are the 'whirls' in which either the electric or magnetic vector is situated entirely in the angular planes. These are called "electropolarised" or "magneto-polarised respectively and are somewhat analogous to the terms "transverse electric" and "transverse magnetic" or "H-waves" or "E- waves" in wave guide theory. In order to develop the above conceptions of polarization, we may write, in the first instance, the expressions G and H, for the electropolarised 'whirls'

$$\beta G_\xi = 0$$

$$G_\rho = i(J_{n+1} + J_{n-1})\; G_\theta$$

$$= J_{n+1} - J_{n-1}$$

$$\beta H_\xi = 2\,2\sqrt{Y^2 + J_n}$$

$$H\rho = i\chi\left(J_{n+1} - J_{n-1}\right)H\theta$$

$$= \chi\left(J_{n+1} - J_{n-1}\right).$$

ρ = radial coordinate of helical system.

$$\text{where} \quad J_n \equiv J_n\left(\rho\sqrt{\chi^2+1}\right)$$

These are Bessel functions of the first kind (Bessel coefficients) of the order V_n of an argument; $J\sqrt{\chi+1}$

which is the same in all cases when $\rho = \rho_0 V$ is an invariant coordinate.

The solutions for the "magneto-polarised whirls" would be expressed in exactly the same way but with the sole difference that the functions $\vec{G}$ AND $\vec{H}$ in the above would be interchanged.

The characteristic expression for the "compound whirl" may now be written;

$$\vec{E} = A\beta\rho_0 R\left(\int_{\chi_1}^{\infty} \frac{f(\chi)}{f(\chi_1)} e^{[i(\tau+n\theta)\mp\chi\xi]}\vec{G}\,d\chi\right)$$

$$\vec{B} = A\beta\rho_0 R\left(\int_{\chi_1}^{\infty} \frac{f(\chi)}{f(\chi_1)} e^{[i(\tau+n\theta)\mp\chi\xi]}\vec{H}\,d\chi\right)$$

where $f(\chi)$ is an arbitrary function insofar as this is consistent with conditions of physical reality.

There is an important case in which;

$$f(\chi) = \exp[\alpha\chi \mp i\chi\alpha_2(\chi)]$$

Here, the second term of the Right Hand expression indicates the dispersion of the centres of elementary "simple whirls" which together constitute the compound one.

When the upper limit of the integral $\left(\int_{\chi_1}^{\infty} \frac{f(\chi)}{f(\chi_1)} \cdots d\chi\right)$

is finite i.e. when χ_L is substituted for ., the type of 'whirl' which would result is defined as a 'fragment.'

This would be a relatively unstable entity.

Properties.

From the above formulation, it should be appreciated that a 'whirl forms waves (in the azimuthal direction) which rotate about the axis with an angular velocity;

$$\Omega_n = \partial/_{\partial t} \big/ \partial/_{\partial \theta} = \frac{\beta\Omega}{n} = \frac{\beta_0 q_0}{n}$$

Hence, in the radial direction, there result standing waves corresponding with the nodal circles of the Bessel functions outlined above, or of their appropriate integrals. In a 'simple whirl' the loci of the nodal circles i.e. the nodal surfaces will be cylinders, in a compound whirl' or 'fragment' they will be surfaces of revolution with curved generating lines.

The total energy of the 'whirl' system may be determined to be;

$$\hat{E} = \frac{1}{8\pi}\int(\vec{E}\cdot\vec{E} + \vec{B}\cdot\vec{B})\,dv$$

(the dots indicate scalar multiplication).
Extended over all space the above quantity becomes;

$$\hat{E} = \frac{1}{8\pi}\int_{-\infty}^{\infty}\int_{0}^{\infty}\int_{0}^{2\pi}(\vec{E}\cdot\vec{E} + \vec{B}\cdot\vec{B})\,dx\,dr\,d\theta$$

The energy of a compound whirl being finite, its existence in a complete form within a confined space is physically possible. On the other hand, a 'simple whirl' or a frafment, for which ___ could only be partially realized in a limited space.

For a compound whirl' where energy is finite,

$$\hat{E} = \beta\cdot\hat{E} = mc^2$$

$$= \beta r_0 c^2 = \hbar c q_0 = \gamma h$$

$$\hbar = \frac{h}{2\pi}$$

$$= 4\frac{A^2}{c}\,\frac{x_1^2+1}{x_1}$$

$$= \Omega\hbar = \frac{\Omega_x n h}{2}$$

$\hbar$ is a constant which has the dimensions of angular momentum which remains invariant for a given 'whirl'. It corresponds to Planck's constant although it might not be equal to it numerically. The angular momentum of the 'whirl' is $2n\hbar$ Thus the invariance of $\hbar$ represents the law of conservation of angular momentum imparted to the system.

The 'Whirl' in a Conducting Medium.

If the electrodynamic 'whirl' system is generated not in vacua but in a conducting medium such as an ionized gas (seeded or unseeded) or a plasma with a resistivity ρ_w, the dielectric constant ϵ both in e.s.u. and permittivity μ in e.m.u.

Then, the electric and magnetic vectors would be expressible in the form;

$$\vec{E} = \frac{c_{11}}{c}\vec{E} \quad ; \quad \vec{B} = \vec{B}_1 + \vec{B}_2$$

and the current density may be expressed in the form; $I = \frac{c_{12}}{c}\,\frac{E_2}{\rho_w}$

$$\text{where} \quad \frac{c_{11}}{c} = \frac{1}{\sqrt{\epsilon\mu}} \quad ; \quad \frac{c_{12}}{c} = \sqrt{\frac{J\Omega\rho_w}{4\pi\mu}} .$$

The expressions for E_1 and B_1 and for E_2 and B_2 will be the same as those previously given for E and B, but with c_{11} and c_{12} substituted.

The resistivity of the air ρ_w depends upon the degree of its ionization and this depends mainly upon $\vec{E}$ which represents the intensity of the electric field. The component $\vec{E}$ is analagous to the local "e.m.f. of ohmic resistance" per unit length, the product of resistivity x current density. Quantitatively $\vec{E}_1$ would be >> than $\vec{E}_2$.
When $\vec{E}_1$ >> 3,5,00 volts/cm complete ionization in most gases results and the condition of plasma is realized.

In order to generate the highly energetic plasmoid systems described a special vortex tube wound with a sheath helix with a geometry conformable with the boundary conditions demanded by the system. Electric power supply would be available as polyphase current and a pumping field acting on the gas would set up a lag between the linear and angular velocities of the gas and the linear and angular velocities of the field. The imbalance resulting from this would favour ionization. Over and above the energy thus expended entrainment would also take place as a result of coupling effects together with a substantial compression of the gaseous core or kernel represented by the 'whirl.'

The total energy output required to ionize F particles under normal conditions would be;

$$e\,F\,v_i \,/\, n_i \qquad \text{per particle.}$$

e is the electric charge
F is the number of particles ionized
v_i is the ionizing potential
n_i is the ionizing efficiency.

The amount of energy transferred to a plasma more fully ionized than that envisaged in the above theory is given by Miller (12) in terms of a propagation constant k.

$$R^2 = \left(\frac{\omega}{c}\right)\left[1-\left(\frac{\omega_{pe}}{\omega}\right)\right]\left(\frac{1}{1+\omega_{e}/\omega+j(\nu/\omega)}\right)$$

where

ω is electromagnetic field frequency,
ω_{pe} is plasma electron frequency.

It should be noted that in communications on applications of 'whirl' theory to deflection of light in the Sun's gravitational field on theories of nuclear structure and of the nature of fundamental particles Japolsky has claimed confirmation by the calculation of fourteen physical constants applying in his formulae only five well known and established factors (c, h, e, m_+ and) the velocity of light, Planck's constant, the electronic charge, the protonic mass and the gravitational constant respectively. His data derived from the above includes the mass ratio of the electron and proton the mass of the neutron and of the light nuclei $_1H^2$, $_1H^3$, and $_2He^4$. It is remarkable to note that these conceptions have permitted a calculation of the period of rotation by pure theory for the planet Venus of 20hrs and 45 minutes. Radioastronomical data published after that gave 22hr, 16minutes. There are, in addition, some interesting and speculative proposals for the realisation of nuclear power from a different route than fission and fusion resulting from theis analysis. Consideration of this, however is outside the scope of this paper.

A certain resemblance may be detected to 'whirl' theory in the interesting work of Dr.Bostick on "Plasmoids.(16,17.). In his experiments which have given rise to some thought-provoking analogies between the behaviour of small slugs of plasma generated by electrostatic means and that of fundamental particles. He has briefly examined the behaviour of a hydromagnetic model of toroidal configuration analagous to an electron or a proton. In his analysis, Bostick has related rest mass energy for the toroidal electron model with its electromagnetic field energy , which he finds to be

$$E_f = \int \frac{1}{8\pi}\left\{|\vec{E}|^2 + |\vec{B}|\right\} dV = e/R\left(1+\frac{1}{\pi}\ln\frac{R}{r_c}\right)$$

V here is volume
R is the major radius
r is the minor radius.

$|\vec{E}| \cdot |\vec{B}|$ are scalar products

This author has pointed out that in any force-free configuration involving plasma or magnetic fields such as those which assume a toroidal form, there are helicoidal enveloping field lines which thread the system such that their pitch angle varies from 90 deg. at the annular axis to approximately zero at the periphery. This led him to postulate self-gravitational forces with such a particle model comparable in kind but not, of course, in degree of magnitude with similar self gravitational effect within force-free electrically conducting plasmas occurring on a galactic scale within the stellar universe.

The problem of designing suitable accelerator and divertor equipment maym at first sight appear formidable particularly when the exceedingly massive nature of most acceleration equipment is visualized even when these are designed to operate under conditions of high vacua. However this type of equipment is not always massive and the work of Budker and Naumov in the Soviet Union may be noted in this connexion.

These workers have published accounts of their development of such devices as an impulsive non-iron (air cored) synchrotron for which a stabilibised beam of relativistic electrons is set up and in which a small addition of ions is needed to compensate for decay of the electron beam through coulomb interaction. With this beam magnetic fields of 10^5 Oersted and electric fields of 10^5 v/cm would result. The beam in question which is described as transparent for ions and for e/m waves and which can be sustained for several hours has been designed specifically to act as a focussing system for the light cyclic resonance accelerator for which Budker has been responsible. In this, the beam can be sustained in circular form using a current of 100,000 ams with a ring radius of 3 metres producing energy of 100 B.ev. One accelerator operating on the principle with a ring radius of 17 has neen able to furnish a beam of 200-300 m.e.v. The installation would have an installed weight of only 150kg. Acceleration is produced by means of R.F. techniques using millimetre waves (2-4 mm). It can thus be seen that extremely high energy particles can be made available using only medium weight equipment. The adaptation of some of the principles used in design and of data gained by such accelerators would be most useful in constructing thrustors for space propulsion. In addition, an open mind will need to be maintained about new methods of realising nuclear energy and it will be the task of the astrophysisists to assist in this field armed with the newest data being acquired through the space programmes.

If the states of matter be categorised as W_a where W_1, W_2, W_3 are solid liquid and gaseous states and W_4 categorised as the plasma state chemical phenomena (W_1, W_2 and W_3) would come within the energy range of say 5 e.v., corresponding to valence bond forces. Radioactive phenomena would be placed at the level of about 10 m.e.v. and cosmic ray phenomena at about 10,000 m.m.e.v. There has been suggested that a fifth state of matter W_5 'nugas' may exist composed of electrons and free nucleons. This would be a common state in galactic processes. It is not easy to rate the energy absorbed in the production of this but it would not, in all probability, be of of a very different order to the quantities associated with fission or fusion processed involving W_1, W_2 and W_3 as energy levels. This would be $5 \sim W_5 < 200$ m.e.v. There is also thought to be a sixth state W_6 possible within the ranges $0.2 < W_6 <$ 4G.e.v. This would contain a good many free mesons, decayed nucleons as well as electrons.

since it is now commonly recognised that upward of 99% of matter in the cosmic universe is within the last two categories, the attainment of energy in new forms and the application of these to interplanetary transport can only be a matter of time and it may be found that neither fission nor fusion processes are particularly suitable to resolution of the problem.

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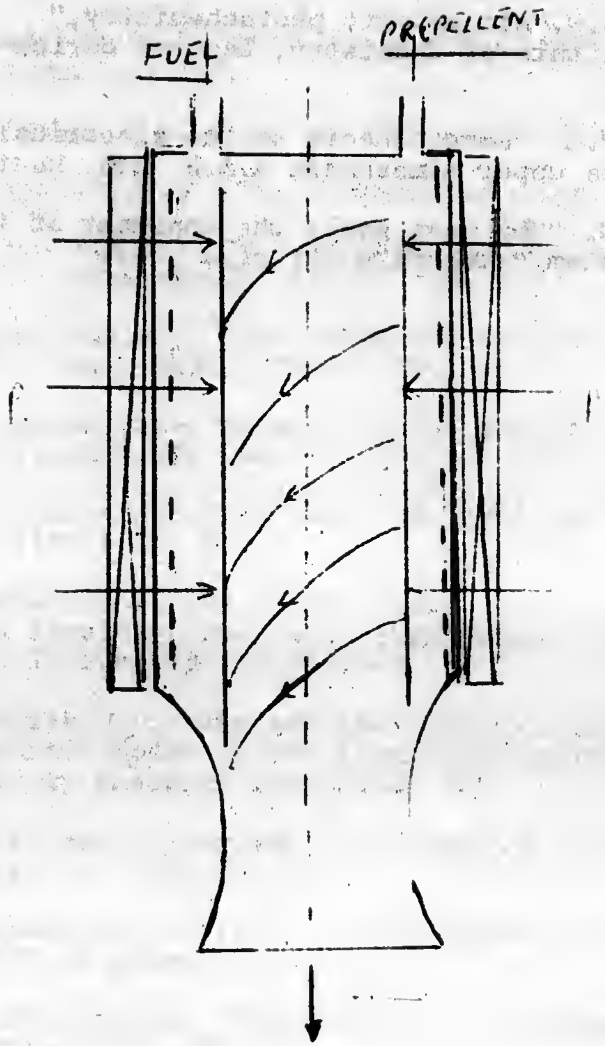
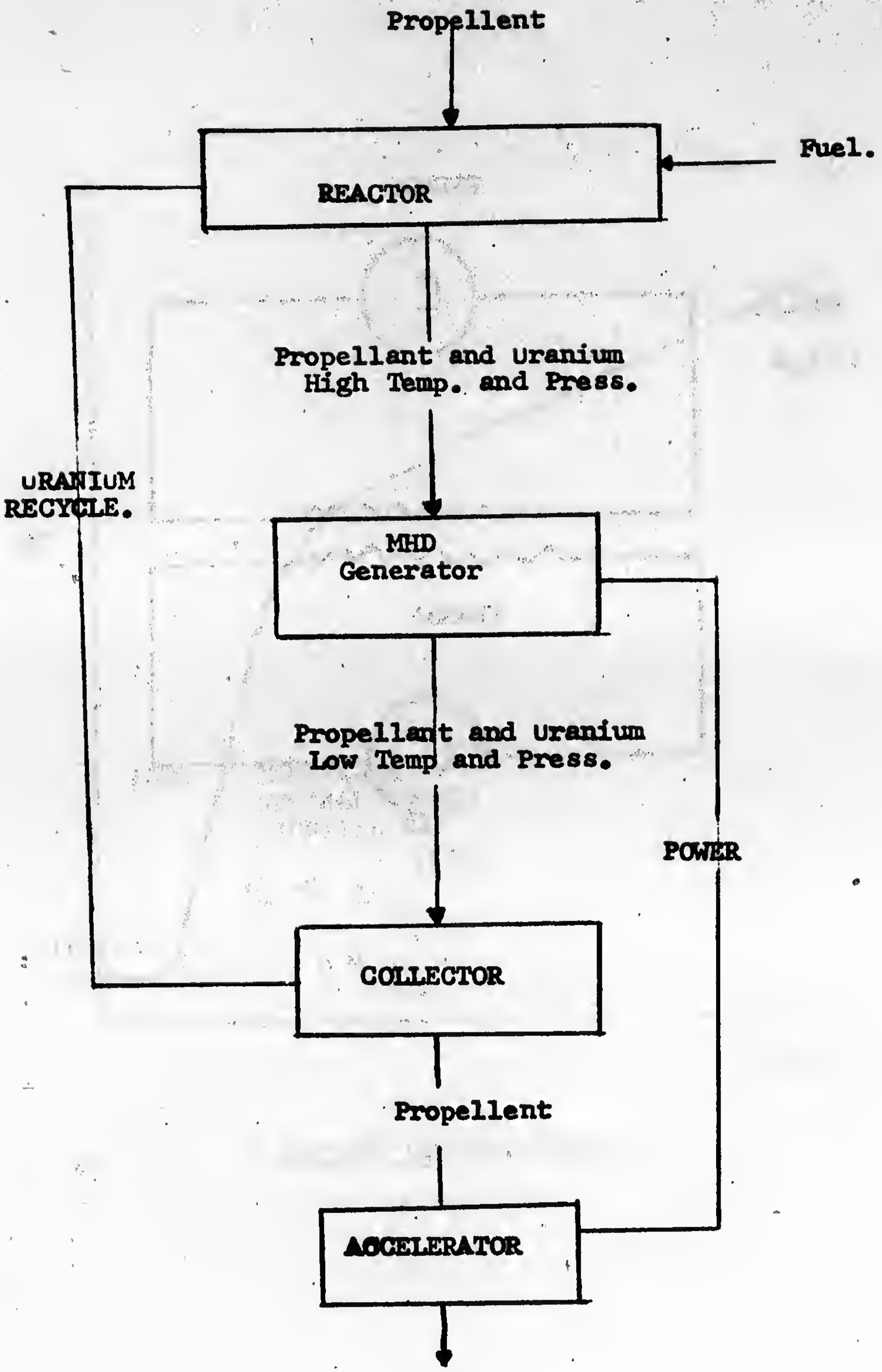
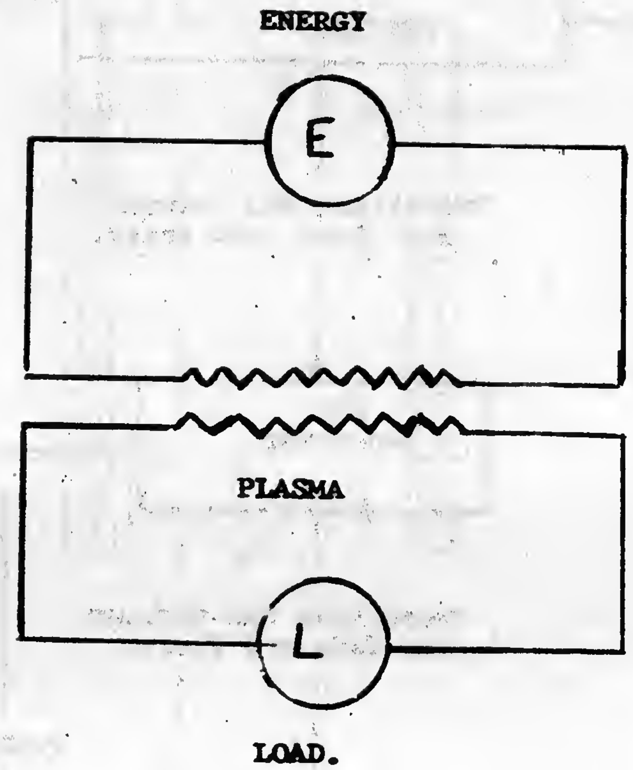


VORTEX CONTAINMENT CAVITY
FISSION REACTOR

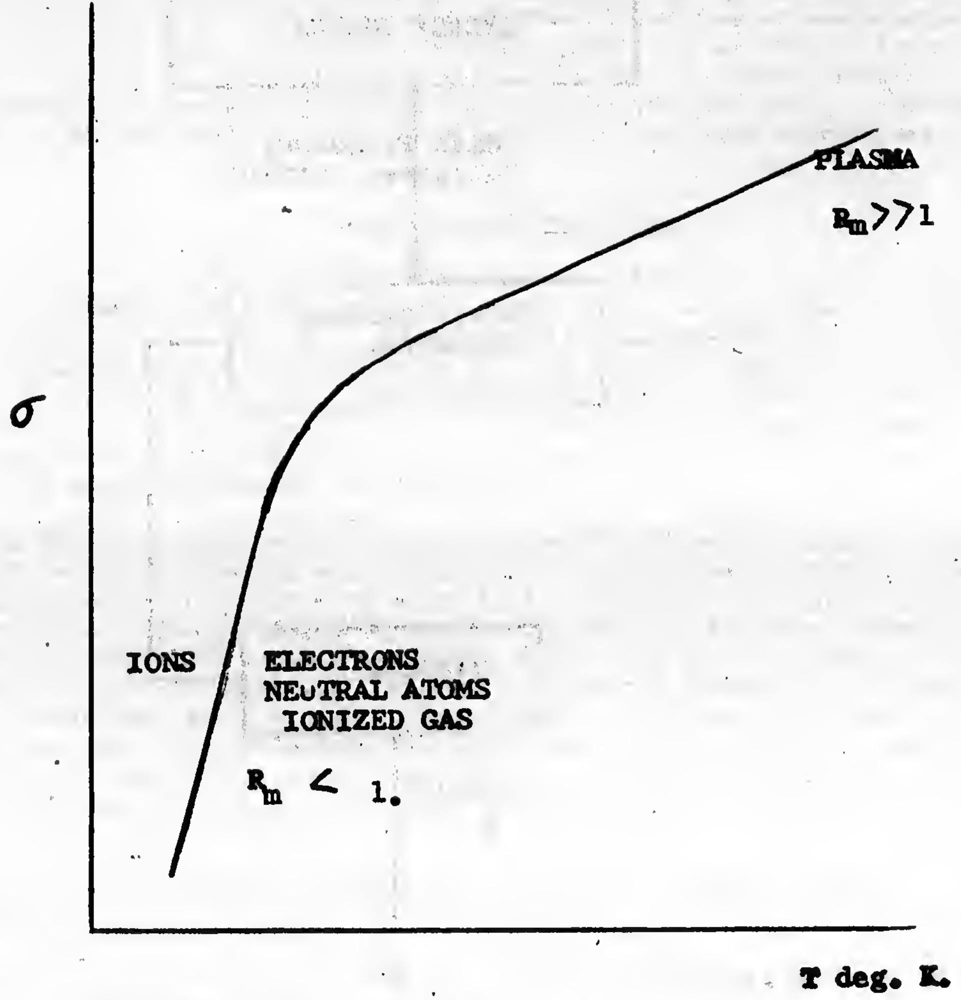
Combined Cavity Reactor and MHD Generator.





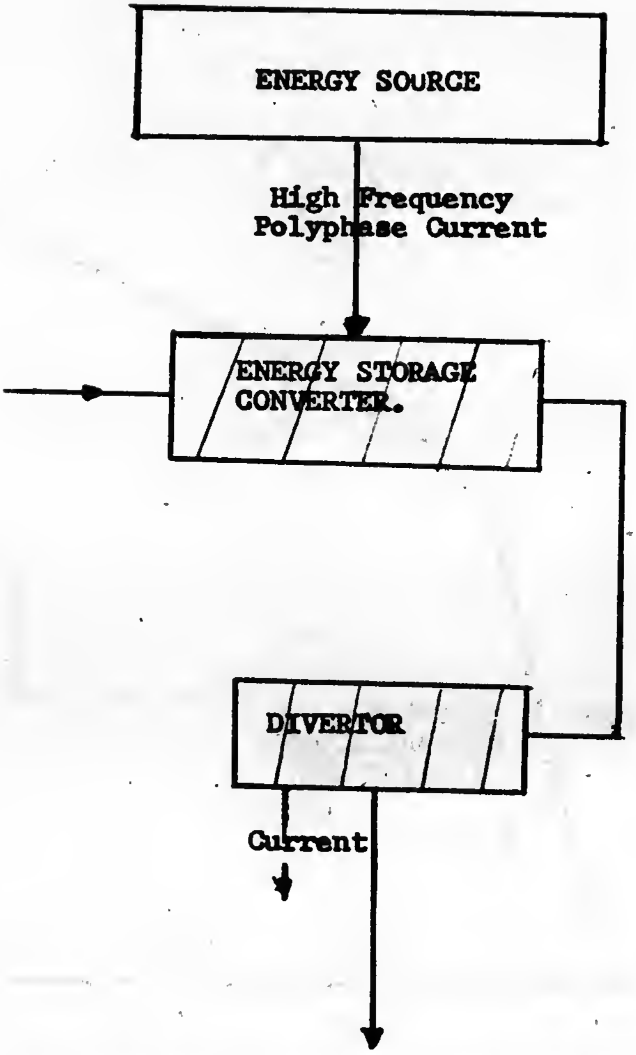
THE PLASMA 'TRANSFORMER'

3.



Regions of Ionization.

4



Storage and Diversion Propulsive System

5.

LABORATORY IN SPACE

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Summary

The manned space station is growing in prominance as the potential next manned space flight project. The NASA plans and studies in the area of advanced manned earth orbital systems are discussed. The requirements for an Orbital Research Laboratory and its uses in the biomedical, scientific and engineering research areas are highlighted. Potential concepts of orbital laboratories ranging from minimum modifications of the Apollo spacecraft to a large 24-man space station are compared.

Background and Objectives

The earth orbital programs of the United States have been, and will continue to be, the foundation for accomplishing the utilization and exploration of space. We have learned from Mercury space flights that man has a potential usefulness in space which may be the basis of achieving new and important benefits in the national interest. As we progress from Mercury to Gemini and Apollo, the U. S. space team will bring us several steps closer to the development of manned space systems that will increase critical scientific and technological skills to ensure our continued growth and international relationships.

In the Apollo Program we will have a spacecraft, three-man crew and ground system that can operate safely and routinely for up to 14 days. NASA is studying the possibilities of an extended duration Apollo spacecraft for early experimentation and research. The Air Force is considering the possibility of extending Gemini with an experiment module to periods of up to 30 days. We believe, however, that the first truly operational Orbital Research Laboratory (ORL) will be a versatile multi-man space station whose flight duration may be as long as several years with periodic crew replacement and resupply. Such a laboratory appears to offer the best means of deriving direct benefit to the scientific community and the general public, and we are carefully investigating several approaches to accomplish this.

There are three promising avenues of approach to the Orbital Research Laboratory open to us:

1. A system based on the Apollo spacecraft with a laboratory section added (AORL) capable of supporting three to six men,

2. A medium sized Orbital Research Laboratory (MORL) with four to eight men, and

3. A large Orbital Research Laboratory (LORL) with 24 men or more.

The choice of which of these approaches or their various combinations is most suitable can only be established by careful study of the various objectives of manned earth orbital flight.

The specific goal of the NASA study effort of manned earth orbital missions is to determine the best methods that can be used to advance space exploration and exploitation through basic research and engineering experimentation in earth orbit. The scientific community at large is expressing a great interest and desire to participate in a sound manned earth orbital research program where one can utilize to the fullest man's ability to observe, discern relationships and exercise judgment. The space systems engineering community is also strongly supporting the establishment of a versatile orbital laboratory to further the development of advanced space flight systems. The early Orbital Research Laboratory would be the forerunner of an operational space platform that provides direct economic benefit to the public, for instance, through greatly improved weather forecasting or even control, earth viewing for agricultural applications, or communications. An orbital laboratory also would play a vital role as a stepping stone to more advanced space operations as a facility for the evaluation of prolonged weightlessness on man's ability to function and survive, and for the assembly and launching of large space ships from earth orbit. Thus the laboratory directly provides the basis for manned exploration of the planets.

There is no doubt that many of the objectives described above can be achieved by means of unmanned satellites; of course, with the exception of those where man himself is a subject. We believe, however, that the complexity required of automatic equipment to accomplish some of them will become rather overwhelming. The eventual earth orbital program will most likely consist of a balance between unmanned satellites and manned orbital space stations. This balance between machine and man depends on the fact that when the sophistication of the mission grows, the machine reliability diminishes, and a point is reached where a man in the system becomes necessary for its success. Besides being able to repair equipment man can be made a very necessary element in the operation of the experiment itself. He can monitor the experiments, calibrate and align equipment, and use his judgment to alter an experiment in progress to meet new objectives. Thus the inclusion of man in the system may be worth tons of electronic gear, which would be required to replace his capability to perform literally thousands of different tasks, accepting thousands of different commands, and providing a reliable on-board logic and memory system.

Because of the large variety of uses to which the ORL can be put it is necessary that it be a highly versatile system. It has to have the volume, weight, power, stabilization and crew capability to perform a large number of tasks and be flexible enough to respond to changing demands. To develop a better understanding of the various demands involved some of the potential uses of an ORL are discussed in the following pages.

Applications

Biomedical Research

One of the initial main uses of the Orbital Research Laboratory would be the study of man's physiological and psychological respons to the space environment and the determination of man's capability for performing useful missions in space over periods of months and years.

Ulitmately we are interested in manned planetary missions which require flight times of over one year. The validation of man's active role in space cannot be accomplished by simulation on earth because of one particular factor peculiar to space:

"Weightlessness." Weightlessness may "decondition" man to such an extent that he would not be able to withstand the re-entry environment on return to earth or the normal gravitational environment after landing. In order to understand the effects of this phenomenon and to determine how to counteract the long-term effects of the space environment on man, all effects on man have to be closely evaluated and compared with earth-based simulation studies.

The space environment may affect, for instance, the following systems:

Cardiovascular System. The maintenance of an adequate cardiac output, with properly oxygenated blood to supply all areas of the body, is vitally important since this is the mechanical transport system for the cellular nutritional needs and waste product removal. The existing knowledge of the role of the gravitational force field, or its absence, on circulatory dynamics is limited. What is known, however, points to the fact that cardiovascular dynamics may be altered significantly to the extent of causing symptoms which would lead to unconsciousness, upon return to a force field after a prolonged period of weightlessness.

Nutritional Functions. The role of a gravity field in the dynamics of food absorption, transport and utilization over prolonged periods of time is unknown at the present time.

Musculo-Skeletal System. The musculo-skeletal system has been subject to speculation as to the role of gravity in maintaining muscle mass, muscle function and strength, and decalcification of bones (mineral mobilization, deposition and balance). It is presently unknown whether these functions are gravity-dependent or primarily dependent upon maintenance of muscular contractions.

Pulmonary Functions. An area of vital importance are the long-term effects of partial pressures of oxygen in excess of sea level partial pressures. What kind of changes, if any, are initiated by an atmospheric composition dissimilar to earth's? How do these possible changes affect such pulmonary functions as ventilation, the mechanics of breathing, and pulmonary ventilation-perfusion ratios during weightlessness? The definition and estimation of the significance of these questions are yet to be determined. Some of these could be duplicated here on earth, but interaction of all multiple factors cannot be predicted.

Biochemical System. Any endeavor which produces stress in excess of measured experience may cause havoc to the endocrine organs. Much needs to be accomplished in understanding the significance of these fluctuations as they relate to the long-term functional usefulness of the individual and his well being after return from the mission.

Psychological Functions. Little is known in the area of psychological variations induced by small closed societies, isolation, and artificial environments, and what is known cannot yet be adequately correlated. Much work must be done so that some adequate measure of assurance can be had that long-term space missions will not pose undue problems in this area.

Vestibular System. This area is related uniquely to artificial gravity produced by rotation and is concerned with the study of the effects of Coriolis and angular acceleration gradients on vestibular functions. Questions which must be answered relate to the degree of interference with function, habituation and effects of changing magnitude of Coriolis and angular velocity as a function of space station configurations.

A biomedical facility in space will require flexibility so that any eventuality can be adequately measured and evaluated. This calls for a high degree of biomedical sophistication. Such sophistication recommends serious consideration of the use of properly trained medical personnel for flight missions.

The final evaluation of all biomedical parameters will lead to one of the key decisions in the future manned space flight program. It will answer the question whether or not an artificial gravity field is required in man's life support environment.

There are three successive objectives in the biomedical experiments:

1. Determine whether man can operate successfully under weightlessness for long periods of time (of the order of one year) and subsequently survive in good condition the environment of re-entry and normal gravity conditions.

2. If there are indications of difficulty in the weightlessness state, determine whether on-board reconditioning measures, such as periodic rides on a low acceleration centrifuge or the occasional applications of pressure cuffs to the limbs, will enable the subject to operate successfully under prolonged weightlessness, followed by re-entry.

3. If there are conclusive indications that periodic reconditioning measures are insufficient, determine the best way of providing artificial gravity by means of rotation of the spacecraft and develop techniques for crew operations in the simulated gravity field.

Scientific Research

While the biomedic research is going on the Orbital Research Laboratory can be also utilized for scientific research. It should be pointed out that some scientific experiment, given high priority today, may be relegated to a lesser rating by the time a space station would become operational. This will depend largely on results of unmanned satellites as well as the military MOL program. However, several typical areas can be identified as being of considerable interest.

Astronomy. An orbiting space station could provide an astronomical observatory that would greatly increase the angular resolution and extend the wave length range beyond that possible in earth-based observatories. However, some experiments on an orbiting observatory might require pointing accuracies as stringent as 0.1 second of arc with photographic exposures for as long as one hour or more. Such stability may be difficult to achieve. A possible solution might be an astronomical mission module in close proximity to the space station with a radio command and an optical data link between the platform and the laboratory with periodic servicing of the platform by a laboratory crew member.

There exists a large range of astronomical observations that may be made from an orbiting observatory. Significant among these are:

1. Ultraviolet, visible and infrared studies of the planets, the solar disk, the solar corona, galaxies, nebulosites and inter-stellar gases.

2. Ultraviolet, visible and infrared studies at very high resolution of stellar systems and a search for planets of nearby stars.

3. Gamma and x-ray telescopy.

4. Radio telescopy.

Only a few basic instruments would be required to do all the studies listed if man were present to align and calibrate the equipment, change attachments to the basic instruments and select the objects to be studied.

Biology. An ORL provides an opportunity for utilizing the unique aspects of the space environment to analyze the relationships that exist between an organism and its environment. Studies would be concerned with environmental effects on photosynthesis, biological rhythms, metabolism, and growth and development of organisms, such as:

1. Study of various physiological systems in the higher animals.

2. Study of embryology.

3. Growth of organisms.

4. Plant physiology.

Geophysics. In this area many useful experiments could be performed. Photography from an orbiting laboratory covering several regions of the spectrum such as infrared and ultraviolet could be used to detect surface features not revealed by conventional photography. With a "multispectral" sensing device the spectral radiation and surface features could be identified and correlated. This would include correlation with known mineralogical composition and major geological features such as volcanoes and moisture patterns, thus developing the techniques for future planetary exploration. From the orbiting laboratory we could also perform panoramic photography of the earth for cartographic purposes with results that would be useful in many scientific fields, particularly meteorology.

Evaluation of the characteristics of electromagnetic matter from high altitudes will make it possible to determine the utility of radar and microwave emission detectors in predicting surface roughness and detailed topography; this may aid in better definition of the earth in oceanic areas by radar altimetry. Detailed definition of the advantages of spaceborne radar systems relative to geoscientific problems would be a major step in determining their future utility in planetary explorations.

Meteorology. With the ability to use high resolution cameras, radar, infrared sensors, and telescopes an orbiting laboratory could contribute significantly in determining meteorological phenomena affecting weather and weather prediction. Specifically the spaceborne laboratory could be used to observe systematically cloud systems as they approach, traverse and leave certain geographic areas with unknown thermal effects in each season. Man's presence would greatly contribute by his observation of peculiar patterns and features and relaying these directly to the ground observers in real time.

Physics and Chemistry. An Orbital Research Laboratory could contribute greatly to investigating the characteristics of space that cannot be measured from the earth. These characteristics, when studied with respect to the overall space environment, electromagnetic and particle activities, may lead to a much greatly increased understanding of materials, communications, relativity effects, artificial radiation belts and planetary system evolution.

Engineering Research and Development

The Orbital Research Laboratory has the potential of providing useful facilities for developing and qualifying the various systems, subsystems, structures, materials and operational techniques that will be required for future manned and unmanned space missions. In addition, an earth orbital space station which could evolve into an orbital launch facility would greatly improve the probability of success for planetary or advanced lunar missions. Launchings from orbit would ease the requirement for boosting large masses to escape velocities by assembling, refueling and checking out in earth orbit planetary exploration vehicles. However, orbital launching will require the solution of a variety of complex engineering development tasks such as assembly in space, checkout, launch procedures, fuel transfer, maintenance, repair, general operations and logistics.

Future systems and materials under development will require long-duration exposure to the space environment. Many parameters of this environment may be simulated in earth-based facilities; however, in many cases large volume and hard vacuum requirements favor in-space testing. Furthermore, ground-based tests obviously cannot provide the conditions of weightlessness or partial gravity for extended periods of time. One should bear in mind, however, that only tests which cannot be effectively carried out on the ground will be assigned to the space laboratory.

Some of the more significant potential engineering experiment groups for Orbital Research Laboratories are identified below.

Crew Systems. The study of advanced life support systems and crew equipment under space conditions would yield valuable information to support man on long-term space missions. The performance and reliability aspects of advanced environmental control systems to provide cooling, heating, pressurization, etc., for future spacecraft configurations can be evaluated. Particular tests would yield data on the effectiveness of leakage detection systems to pinpoint micrometeoroid penetration and failures in hermetic seals. This, in turn, would lead to an assessment of sealing techniques to correct these penetrations and failures.

Extravehicular Operations. For extended extravehicular operations man must be well versed in the capabilities and limitations of individual propulsion units, extravehicular suits and portable life support systems. The Orbital Research Laboratory would provide a test bed to train personnel for future lunar, earth orbital and planetary missions.

Electrical and Electronic Systems. The performance and/or endurance capability of various solar and chemical power generation systems operating in space environment can be evaluated. Various items of communication equipment, such as very high frequency devices, may be tested to develop improved space communications in terms of advanced components, optimum frequencies, bandwidths, transmission power, receiver sensitivity, reliability, etc. The performance characteristics of long lifetime advanced navigation and control systems can be studied and new guidance techniques evaluated.

Propulsion Systems. The effects of the space environment, especially weightlessness and long-time space exposure on ignition devices, re-start capability, fuel sloshing, vortexing and expulsion could be determined. Tests could be conducted on various fuels and oxidizers in different propellant tank configurations and with various insulation techniques and materials. Electric propulsion systems could be tested under true space conditions.

Structures and Materials. Many simultaneous effects of the space environment such as meteoroids, vacuum, radiation and temperature cycling on various characteristics of materials and structures are difficult to be meaningfully reproduced in ground tests. Ablative materials could be exposed to space vacuum and temperature extremes for long periods of time to evaluate their integrity. The zero gravity environment may be suitable for the manufacture of new materials. An effective meteoroid protection research program could be undertaken with the aid of trained experimenters who would help deploy large meteoroid bumper areas, examine penetrations and alter the materials to devise optimum protection schemes.

Development Flight Testing. Also of interest is flight testing of large unmanned satellites and space probes. Present satellite projects require a costly and time consuming development flight test program necessary to obtain reliable flight hardware. This is mainly due to the fact that we have not yet learned to build adequate ground simulators, and, thus, testing of final satellite configurations on the ground does not allow an isolation of all design deficiencies. By orbiting the test article as a "tag-along" near the orbital laboratory, such as a large deployable balloon or antenna system, thus allowing visual observation and a short command and telemetry link, diagnosis and repair of satellite malfunctions would be possible through extravehicular operations of the crew.

Laboratory Concepts

NASA, coordinating fully with DOD, has been studying the feasibility of manned Orbital Research Laboratories and the experimental requirements placed on such spacecraft. These study efforts, which are continuing, are conducted in-house NASA as well as through contract with the aerospace industry. The investigations have yielded to date a significant amount of data regarding many phases and aspects of potential advanced manned earth orbital systems. They have explored on a preliminary basis the modifications necessary to hardware presently under development within the Apollo program so that the same basic spacecraft could be utilized for the earth orbital missions, and the studies have looked at new vehicles that could accommodate from four to twenty-four men. Four different types of orbital systems are being considered:

1. Extended Apollo.

2. Apollo Orbital Research Laboratory (AORL).

3. Medium Orbital Research Laboratory (MORL).

4. Large Orbital Research Laboratory (LORL).

The first three are designed primarily for a Saturn IB launch vehicle. The large ORL utilizes the Saturn V capability. For initial missions these laboratories are all considered to be launched into circular earth orbits at altitudes of 200-260 miles and inclinations of about 28 degrees.

While examining the means by which a manned orbital laboratory capability could be realized, the studies also continue to investigate logistics systems that would support these orbital laboratories and explore the operational requirements of such systems. The Gemini B/MOL is being examined as a potential integral element of the ORL program. On the basis of these studies we will be able to compare the various alternatives and arrive at a choice of a system that is most suitable from the technical as well as economical point of view. It is becoming increasingly clear, however, that Extended Apollo is an essential element of an expanding earth orbital program. In the initial stages it would be used as a laboratory, and later it could be converted to a logistics system. The AORL, MORL and LORL, on the other hand, are presently competitive systems, and a decision will have to be made which is the preferable first generation ORL system.

Some of the study results to date are highlighted below.

Extended Apollo

Several methods for extending the capability of the Apollo spacecraft for longer earth orbital missions are being explored. Although currently designed for a fourteen-day mission to the moon and back, preliminary feasibility studies indicate that the mission life time capability of the spacecraft could be extended to periods of up to three months without major modifications.

In this version (Figure 1) only supplies such as life support stores and stabilization propellant are added, and some redundancy is provided in the subsystems. The basic Apollo subsystems are not replaced by more advanced equipment, thus allowing an uninterrupted growth of Apollo from its presently planned earth orbital tests. The environment control system is only changed to the extent of providing a two gas atmosphere (adding nitrogen), because pure oxygen may be harmful for periods of time over 30 days.

The major limitation of the Extended Apollo as a laboratory is its limited pressurized volume (360 cubic feet). By eliminating one crew member and his restraint system there can be made available for experimentation about 190 cubic feet of volume in the command module of a 45-day Extended Apollo. The removal of lunar mission propellant and tankage from the service module provides then an additional 1,000 cubic feet of volume for experimental payloads. With a Saturn IB launch the experimental payload available at an altitude of 150-200 nautical miles is about 5,000 pounds.

The Extended Apollo capability can grow to over 120 days when some subsystems are changed; for instance, one of the required changes would be the replacement of the present fuel cell power plant by a solar cell system.

Apollo Orbital Research Laboratory

The AORL is an outgrowth of the Extended Apollo, which incorporates a 5,600-cubic foot pressurized laboratory module in the adapter between the service module and the S-IVB booster stage made vacant by the removal of the Lunar Excursion Module (Figure 2). Such a laboratory would be launched initially with a crew of three in the Apollo command module. In orbit the Apollo would disengage from the laboratory, turn around and dock nose first to the airlock, thus allowing the crew to enter the laboratory. Another Apollo spacecraft could ferry up three additional crew members and supplies, which would dock at the opposite end of the laboratory. From then on for at least a year, on a three-month cycle, new Apollo ferries would deliver fresh crews and supplies or equipment. It is also possible that modified Gemini-B/MOL spacecraft may be used as ferries and modules carrying experimental gear.

Medium Orbital Research Laboratory

The MORL is directly comparable to the AORL, except that it is launched unmanned. Without the necessity of carrying the Apollo spacecraft the Saturn IB can deliver a heavier laboratory into the 200 nautical mile altitude circular orbit. Like the AORL it accommodates a six-man crew which is launched on subsequent ferry flights. The additional weight margin over AORL allows for such things as carrying of more elaborate equipment, providing better radiation protection, and a hangar for docking, repair and unloading of ferry vehicles. Figure 3 shows a Gemini ferry in the hangar. Apollo or Gemini B/MOL ferries could be also accommodated.

The basic laboratory, as presently conceived, is a sphere, twenty-two feet in diameter surrounded by a cylinder to make its aerodynamic shape compatible with the Saturn IB launch vehicle. The pressurized living volume is 5700 cubic feet with 2,800 cubic feet devoted to experimental usage.

The design shown has two compartments, the upper equipped as living quarters and the lower as a laboratory. Between decks there is a centrifuge provided for crew reconditioning and biomedical research. Four kilowatts of electrical power are supplied to the laboratory by solar cell panels. The internal atmosphere is maintained by reprocessing the air to recover oxygen from carbon dioxide. Water is reclaimed from metabolic wastes.

Like the AORL the MORL is basically a zero gravity station, but could be conceived in such a way that it could be adapted to provide an artificial gravity field for the crew in the laboratory. The laboratory and the expended upper stage of the launch vehicle could be rotated about their common center of mass utilizing a connecting system of cables (Figure 4). Note the stored ferry vehicles on the side of the laboratory. These vehicles, of course, have to be stored at the station since the crew inside the laboratory must have means by which to abandon the laboratory in case of a disaster.

Large Orbital Research Laboratory

The LORL is characterized by a crew of twenty-four or more. It would be launched with a two-stage Saturn V vehicle into a 260 nautical mile orbit, and maintained there for a period of up to five years. Figure 5 shows one of the concepts presently under active consideration. It is a three-radial-module design with a diameter of about 150 feet. It includes a zero-gravity laboratory at the central hub and radial modules which can be rotated to provide artificial gravity. Various individual sub-laboratories can be accommodated in the radial modules at varying levels of gravity. The LORL has a total volume of about 67,000 cubic feet and weighs about 247,000 pounds.

The long mission duration of the LORL requires a very efficient life support system. Like in the smaller laboratories the atmosphere is provided by regenerating oxygen from carbon dioxide, and water is maintained by reclamation from urine and waste water. Thirty kilowatts of power are provided by approximately 7,000 square feet of solar cells. The crew and supplies are transferred between the earth and the space station by means of Apollo like six-or twelve-man ballistic or twelve-man lifting body ferries. Figure 5 shows a lifting body ferry docked at the station.

The LORL offers a flexibility of performance, manpower, space, and electrical power for a wide range of experiments which the smaller laboratories cannot provide. It could possibly be adaptable to operational growth into an orbital launch facility for planetary and advanced lunar missions. This great versatility may provide greatest cost effectiveness in spite of the initial high investment.

Operations and Logistics

The realization of any of the Orbital Research Laboratories discussed here will require an operational support system that will heavily tax our present capability in this area. Presently and in the foreseeable future space flight operations will utilize the launch facilities and global tracking networks on an intermittent basis with each manned launch receiving very special treatment. The gradual evolution of an orbital laboratory system will require these operations to be performed on a much more routine basis. To assure maximum utilization of the present facilities and those planned for the Apollo program the operations associated with the orbital laboratory will have to be conceived to be as simple and similar to other operations performed in connection with manned space flight as possible. Interactions with other programs will have to be defined precisely and flexibly in order to avoid delays due to coincident facility utilization at the launch site or the control center. Regardless of the method of ORL launch, ferry and resupply operations will be required for any extended duration missions. These operations will be necessary to replace the crew, provide life support and stabilization expendables, replace experiments, deliver spare parts and provide fuel for orbitkeeping or changing. The requirements on the ferry/logistics vehicle are that it must have a round trip capability and always be ready at the ORL for emergency evacuation. This in turn enforces severe requirements on the spacecraft in terms of long-term orbital storage capability as well as capability for rapid checkout and maintenance.

Crew replacement could be accomplished by a ferry spacecraft using launch, rendezvous, re-entry, landing procedures, operations and landing sites currently planned and being developed for the Gemini or Apollo program. The choice between Gemini, modified Gemini B/MOL, Apollo or a new generation of larger ballistics or lifting body ferry vehicles and their respective boosters is presently unresolved and will depend upon the operational requirements for crew replacement and resupply. Figure 6 shows the various ferry vehicles and laboratories presently being considered.

A great amount of study is still required to illuminate all of the systems possibilities to determine technological development requirements and to establish a coherent earth orbital program. Only through a concentrated effort on the part of both NASA and DOD to recognize each other's individual and our mutual goals can we arrive at an economic solution to the Laboratory in Space.

Figure 1. EXTENDED APOLLO

Figure 2. APOLLO ORBITAL RESEARCH LABORATORY

Figure 3. MEDIUM ORBITAL RESEARCH LABORATORY

Figure 4. ARTIFICIAL GRAVITY PRODUCED BY ROTATION

Figure 5. LARGE ORBITAL LABORATORY

Figure 6. LABORATORIES AND FERRY VEHICLES

BIOMEDICAL REQUIREMENTS FOR SPACE CABIN ENVIRONMENTS

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Introduction

In this paper I plan to discuss the physiological needs of space crews that must be provided for by the life support systems of space craft. I will make no effort to define how the space crew's physiological needs will be met. This is a task for the designers of life support systems, one involving all of the scientific disciplines and a subject far too broad and technical for me, as a physician, to attempt. I can only hope to present the more urgent physiological requirements of space crews for which provision must be made when man travels in space.

I have divided these physiological requirements into several categories, each of which will be discussed independently. Listed in order of decreasing priority, the space crew will require:

1. A livable atmosphere.
2. Food and water.
3. Radiation protection.
4. Provision for personal hygiene.
5. Provision for control of fatigue.
6. Protection from the effects of weightlessness.

For even the shortest space mission, the first of these, a livable atmosphere, is absolutely essential. The others assume increasing importance with increased length and complexity of the space mission.

A Livable Atmosphere. The ideal atmosphere would be that which exists in an air-conditioned building at sea level in which the individual can select the most comfortable temperature and humidity to suit himself. However, with proper clothing and some acclimatization, man can adapt himself to rather extreme atmospheric conditions. Man can, and does, live in the desert, the jungle and the arctic. He can live at sea level or at altitudes of two to three miles above sea level, yet there are uninhabitable areas on earth as a result of temperature and altitude extremes to which man is unable to adapt. Somewhere between the extremes of temperature, humidity and atmospheric pressure that exist in the desert, jungle or mountains, and that of the air-conditioned sea level building, we will find the conditions in our space cabin. Our problem is to determine the most ideal atmosphere for the space cabin.

The ideal atmosphere for man on earth is not necessarily the best atmosphere for use in space cabins. I would suggest we look first at the minimum requirements a space cabin atmosphere must provide for, if man is to survive in space, and then see what can be added to this minimum atmosphere to improve upon it.

The single most important function of man's atmosphere is the provision of oxygen for respiration. For oxygen to be useful to man it must be of sufficient partial pressure when it reaches the alveoli of the lungs so that it is picked up by hemoglobin and circulated to body tissues. It is desirable that hemoglobin be 100 per cent saturated with oxygen as the blood circulates through the capillaries of the lungs. The minimum partial pressure of oxygen required in the alveoli of the lungs to provide for 100 per cent saturation of hemoglobin is approximately 100 mm of mercury. Two other constituents are always present in alveolar gas: water vapor and carbon dioxide. For this reason, breathing 100 per cent oxygen at 100 mm of mercury, pressure will not provide for 100 per cent saturation of hemoglobin with oxygen. Normally, depending upon the atmosphere breathed, between 30 and 40 mm of carbon dioxide is always present in the alveoli and 47 mm of water vapor, the vapor pressure of water at body temperature. The partial pressure of carbon dioxide and water vapor in the alveoli is normally about 87 mm of mercury. Therefore, to provide a partial pressure of oxygen of 100 mm of mercury in the alveoli, the total pressure must be 187 mm in the alveoli and, thus, 100 per cent oxygen at 187 mm of pressure must be inhaled to provide for 100 per cent saturation of hemoglobin. This is roughly the environment to which todays crews are exposed when wearing full pressure suits, a 100 per cent oxygen environment at 3.5 psi or 183 mm of mercury.

At this point it might be best, from the standpoint of clarity, to change units. Conventionally, when discussing respired gases, the partial pressures of gases are measured in mm of mercury. When discussing cabin pressures, the most commonly used unit is pounds per square inch.

As just mentioned, the minimum atmosphere that can be used which will maintain space crews' oxygenation at 100 per cent is 3.5 psi, the pressure used in todays pressure suits. In order to provide a margin of safety in the event of a cabin leak or puncture, todays space craft, Mercury and Gemini, were designed to operate at 5 psi. For the shorter space missions, the use of 100 per cent oxygen at 5 psi has proven quite satisfactory.

There are two good reasons for not selecting cabin pressurizations of less than 5 psi. First, as already mentioned, the use of lower pressures would be quite hazardous in the event of an undetected cabin leak or puncture. If a minimum pressure of 3.5 psi were used, for example, and a puncture occurred, the space crew might become hypoxic too rapidly to take corrective action. Another problem with selecting cabin pressures below 5 psi is related to bends. Whenever air crews are pressurized below 5 psi, bends occur quite frequently. However, at pressurization above 5 psi, bends are extremely rare. Pre-breathing, prior to launch, of 100 per cent oxygen for several hours can greatly reduce the incident of bends following depressurization to below 5 psi, but does not completely eliminate this possibility. I feel that both of these factors were important considerations in the solution of a cabin pressure of 5 psi for Mercury and Gemini.

Using a single gas system, that is 100 per cent oxygen, it is impractical to use a pressure of more than 5 psi because of the problem of oxygen toxicity. For prolonged missions there may even be a problem with oxygen toxicity with the 5 psi 100 per cent oxygen system. Low pressure chamber runs with such an atmosphere have indicated there may be some minor problems after exposure of two weeks or more. Low pressure chamber studies have been conducted with the 5 psi 100 per cent oxygen atmosphere by the Air Force, Navy and Republic Aviation.[1] In all of these studies there have been some indications of oxygen toxicity, as manifested by eye irritation, coughing, substernal pain and aural atelectasis. In the Republic Aviation studies, there were some hematological and urinary changes which are requiring further investigation. In the Navy studies, some change in peripheral vision has been noted during night adaptation after subjects were returned to sea level. However, it has not been felt that any of these problems are of sufficient magnitude as to prevent the use of the 5 psi 100 per cent oxygen atmosphere for missions of up to two weeks in length. For longer missions, further study is required to determine the suitability of this atmosphere.

Another serious problem with the use of the 5 psi 100 per cent oxygen atmosphere is the increased fire hazard. Although there have been no fires aboard the Mercury space craft, two fires have occurred in altitude chambers using this atmosphere, one at the Air Force School of Aerospace Medicine and one at the Navy Aircrew Equipment Laboratory. A study at the Naval School of Aviation Medicine has shown that in this atmosphere, paper ignites at a lower temperature and burns approximately six times faster than in our normal sea level atmosphere.[2] It was also found that neoprene coated nylon twill, lightweight nylon and vinyl plastic all ignited in this atmosphere, whereas, with the normal sea level atmosphere, these materials melt but do not ignite.

For prolonged space missions, it appears that, from the point of view of the space crew's well-being, our sea level atmosphere would be best. Such an atmosphere gets around the oxygen toxicity problems and drastically reduces the fire hazard. At the same time, a new hazard is introduced, the problem of bends in the event of loss of cabin pressure. This does not necessarily mean that a sea level atmosphere is still not most desirable from a physiological standpoint. We can get around the bends problem by developing pressure suits that operate at 5 psi or more. Even so, the sea atmosphere is not the most practical. Healthy individuals function very well in cities like Denver, Colorado, where the ambient pressure is about 12 psi. For that matter, many of us have camped in the mountains on hunting and fishing excursions at altitudes of 10,000 feet where the ambient pressure is down to 10 psi. At this altitude, breathing our normal earth atmosphere, the alveolar partial pressure of oxygen drops to about 61 mm of mercury but, because of the affinity of hemoglobin for oxygen, our arterial oxygen situation remains at about 90 per cent. Such an altitude is tolerated quite well by healthy individuals, the only measurable effect being some loss of dark adaptation by the retina of the eye and, thus, a loss of night vision. A slight enrichment of the atmosphere with oxygen can easily correct this. Such an atmosphere would greatly reduce the likelihood of bends in the event cabin pressure is lost and the 3.5 psi pressure suit must be relied upon. Also, the fire hazard is much lower than with the 5 psi 100 per cent oxygen system. I feel rather safe in predicting that man could function at 100 per cent efficiency for an indefinite period of time in space with a 30 per cent oxygen, 70 per cent nitrogen atmosphere at 10 psi. Probably, an atmosphere of 50 per cent oxygen, 50 per cent nitrogen at 7 psi could be tolerated for an indefinite period of time, but more work needs to be done on such atmospheres in altitude chambers before this can be said with impunity. I think it highly doubtful that the present 5 psi, 100 per cent oxygen system will suffice for prolonged space missions.

For a more detailed approach to the selection of space cabin atmospheres, I would refer you to the article in the August 1963 issue of Astronautics and Aerospace Engineering by Parker and Ekberg.[3] However, before leaving the subject of a livable atmosphere and moving on to the other aspects of life support systems, some mention of carbon dioxide management, control of toxic materials and the control of temperature and humidity must be made.

The present threshold limit value for carbon dioxide, as recommended by the American Conference of Governmental Industrial Hygienists, is 5000 parts per million. This is based on exposure at sea level, 8 hours per day, 40 hours per week. This corresponds to a partial pressure of carbon dioxide in respired gases of 3.8 mm of mercury. This is a safe level at which no symptoms would be expected. The anesthetic level for carbon dioxide is about 75 mm of mercury. At levels of 20 to 25 mm of mercury, corresponding to an early submarine level, symptoms have been described of a biphasic excitation-depression reaction in humans.[4] The minimum level at which symptoms of carbon dioxide toxicity might be expected lies somewhere between the 3.8 mm partial pressure, as recommended in the threshold limit values, and the 20 to 25 mm level. I would suggest that, for planning purposes, one should strive for the 3.8 mm level or less.

For other contaminates of the space cabin atmosphere, I would suggest that the threshold limit values for toxic substances, as recommended by the American Conference of Governmental Industrial Hygienists, be used as a guide, keeping in mind that these levels are based on the 40 hour work week and not the 168 hour week to which astronauts will be exposed. As in the nuclear submarine program, every effort must be made to keep all contaminates out of the space cabin atmosphere. When contaminates are unavoidable, they must be identified and dealt with on an individual basis, keeping in mind that two or more contaminates might act synergistically.

The remaining variable, so far as a livable atmosphere is concerned, is that of temperature and humidity. In order for the space crew to maintain heat balance, it will be necessary for them to exercise some degree of control over the temperature and humidity of their atmosphere. Maintaining heat balance is a function of heat gain vs. heat loss by the body. The more important sources of heat gain are metabolic activity and absorption of infrared radiation from the surroundings. Heat loss is primarily controlled by the evaporation of water and the irradiation of infrared radiation by the body. The rate of heat loss by the evaporation of water is dependent upon the temperature and humidity of the air and the rate of circulation of air over the skin. With this many variables to consider, it is impossible to choose any one temperature and humidity standard for the space cabin atmosphere. Heat loss or gain by radiation will be dependent upon the clothing worn by the space crew and the effective temperature of the walls of the space cabin. Heat gain from metabolism will depend upon the activity of the space crew. The amount of clothing worn, the heat load on the space cabin and the activity of the crew will all vary over rather wide ranges during prolonged space flight. The life support engineers must consider all of these factors, in light of the particular mission, in their design of the degree of control the space crew must have over the temperature and humidity of the space cabin.

Food and Water. After provision of the space crew with a livable atmosphere, the next most important biomedical requirement will be that for food and water. I feel that the most important factor to consider in provision of a diet for space crews is that of palatability. The food must be appetizing and palatable, or it will not be eaten. This is important, not only from a nutritional standpoint but also it is an important morale factor. This will be especially true for the longer space missions. Aside from providing space crews with food that is sufficiently palatable to be eaten, other prime factors to be considered are providing adequate nutrition and avoidance of foodstuffs that might cause gastrointestinal disturbances such as diarrhea or constipation. The food must also be provided in such a manner that it requires a minimum of storage space, will not spoil, is readily prepared for consumption and can be consumed under weightless conditions.

So far as content is concerned, the space diet must provide for a total calorie intake of between 2500 and 3000 calories. It should be a high protein, low bulk, low residue diet, containing all of the essential amino acids, fats, minerals and vitamins. Foodstuffs that are diuretic or of high cellulose content must be avoided in order to keep the space crews' output of urine and feces at a nominal value. The problem of urination or defecation while wearing a pressure suit is still a major one with no easy solution. Not only is there a problem in removing and donning of pressure suits, but also one of collection and disposition of these body wastes in the cramped quarters that can be provided the space crew.

The requirement for water will depend on a number of factors, such as the loss of water from the body by evaporation, perspiration, urination and defecation, and the water gained by the body from that contained in foodstuffs and from the metabolism of food. It is estimated that space crews will require from five to ten pounds of water per day for drinking and reconstitution of foodstuffs. The lower value is for the shirt-sleeve environment where heat balance is controlled primarily by control of the environmental temperature. The higher value, ten pounds per man per day, is an estimate of the water requirement when wearing pressure suits where heat balance is maintained by evaporation of perspiration.[5]

Radiation Protection. I have listed radiation protection as next in priority after provision for a livable atmosphere and for food and water. This is not a very critical consideration for earth orbital missions, but is a factor of considerable importance for moon probes and especially deep space probes. It is outside the scope of this presentation to discuss the amounts and types of ionizing radiation that might be encountered on a space mission, or how much of what kind of shielding will be required. Rather, it is my purpose to outline the limits to which crews might be exposed and the risks entailed by such exposure.

At present, the limits for exposure to whole body ionizing radiation for workers in industry where radiation exposure is a hazard is set at 5 rem per year. Such an exposure entails no measurable risk. Approximately the lowest level of exposure to ionizing radiation at which some measurable effect can be detected in man is about 50 rem. Clinically, about all that can be detected following an exposure to 50 rem are some slight changes in the morphology of the cellular elements of the blood. Even this is only a transitory effect. However, statistically, it is estimated that this level of exposure to the entire population would approximately double the incidence of genetic mutations and of leukemia. For the crew of a space craft, this is a rather small risk in comparison to other risks entailed, on deep space probes. I would think that designing shielding to prevent an acute exposure in excess of 50 rem in the event of an unpredicted solar flare would not be unreasonable. This is less than half the dose of whole body ionizing radiation that might produce symptoms of radiation sickness. It would require an acute exposure to something on the order of 150 to 200 rem to produce any symptoms of radiation sickness among the members of a space crew.

Another type of electromagnetic irradiation from which crews must be protected are the direct rays of the sun. Provision must be made to prevent crews from inadvertently looking into the sun. Also, adequate protection must be provided from the intense infrared radiation to which crews will be exposed during extravehicular activities in space.

Provision for Personal Hygiene. I have listed next, in order of importance so far as biomedical requirements are concerned, the provision for personal hygiene. For prolonged space missions, provision for personal hygiene is important as both a biomedical and morale factor. This entails provision for elimination of urine and feces, cleansing of the skin, shaving and cleansing of the teeth. For missions of a few weeks, I suppose the crew can get by without haircuts, but for extremely long missions some provision may even be necessary for haircutting.

For short missions, up to two days, the crew can manage without removal of the pressure suit. However, for anything longer than this, provision must be made for at least partial removal or opening of the pressure suit for purposes of urination, defecation and cleansing of the skin. For the long range missions, more than two weeks, provision must be made for complete removal of the pressure suit and for a shirt-sleeved environment. We must also provide for changing into clean clothing. If we are to prevent skin disease, it is just as important to have clean clothing as it is to bathe.

In providing for personal hygiene, the importance of preventing contamination of the cabin atmosphere should be emphasized, especially in reference to such details as whiskers from shaving and/or materials used in cleaning the skin that might be toxic in recirculated air.

Provision for Control of Fatigue. For shorter space missions, control of fatigue has not been a serious problem. Astronaut Gordon Cooper reported having no difficulty in sleeping during the MA-9 flight.[6] At this point in time, I don't feel that we can say much about sleep during space flight, other than that it is possible to sleep while strapped in one's seat wearing a pressure suit during the weightless state. Whether adequate sleep can be obtained in this manner to prevent fatigue as a result of inadequate rest over a period of days is still a matter of conjecture. The programmed two week Gemini missions should prove very enlightening regarding the requirements for sleep over prolonged periods of space flight.

Fatigue is a difficult thing to measure. Sleep, recreation and work are all factors influencing fatigue. For certain critical phases of space flight it is important that crews are alert and at their peak, so far as performance goes. To insure that this is so, it will be necessary to work out cycles for work, rest, recreation and sleep. However, since it is impossible to simulate the conditions of space flight, especially the weightless state and the anxiety which influences all of these to an unknown degree, we must wait for the longer Gemini flights before defining the requirement for rest, recreation and sleep during longer missions.

Protection from the Effects of Weightlessness. I have listed as the last of the biomedical requirements for space cabin environments, protection from the effects of weightlessness. Before the first orbital flight, there was considerable speculation regarding the possible adverse effects that might result from prolonged weightlessness. However, at least for these shorter flights, exposure to the weightless state has had no ill effects. For longer flights, one might still speculate that there may be undesirable effects. There may be some problem as a result of disuse of certain muscles, the lack of weight bearing on the skeletal system, or the lack of stimulation of the proprioceptive reflexes for control of balance. However, if such effects should result from longer exposure to the weightless state, I feel certain that exercises can be utilized to prevent most, if not all, of these ill effects.

Conclusion

In conclusion, I have attempted to outline the biomedical requirements for the space cabin environment. I have discussed the more important factors bearing on the selection of the cabin atmosphere, food and water requirements, protection from radiation, provision for personal hygiene, control of fatigue and the effects of weightlessness. The satisfactory solution to the many problems entailed in providing for these biomedical requirements for prolonged space flight is a considerable task, but one that the designers of life support systems are accomplishing in an admirable manner.

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ELECTRONIC SLEEP CONTROL IN ASTRONAUTICS

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OUTLINE

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SUMMARY

Astronautic mission profiles require for the crew, various sleep-wakefulness cycles which are quite different from the normal diurnal rhythm. It has been found that there are means by which certain levels of sleep can be induced electronically without undesirable side effects. The paper describes the efforts made so far in the international scientific community and also the results of his own experiments.

The technique of the so-called "electronarcosis" which has been used for a number of years is also described in the paper, whereby the differences of the effects of higher and lower frequencies are described.

The paper also discusses the possible hazards of the application of electric currents to the brain.

I. INTRODUCTION

It is of major concern for the systems analyst of space vehicles to establish the criteria of operational integrity of the man-machine system.

One of the great problems in putting the human element into the mechanism is its endogenous cyclic behavior. Among many inherited and acquired periodicities the rest-wakefulness cycle requires prime attention.

It became obvious from task analyses that the terrestrial indoctrination of eight hours of sleep and sixteen hours of wakefulness will not be possible to maintain during certain types of space missions. Yet, this habit is extremely firmly indoctrinated. The acquisition of this behavioral pattern dates back almost to the imprint stage because it was manifest in the mother. The infant has early acknowledged that any interference with this pattern met with the covert or overt hostility in its environment. This early adjustment was reenforced later by the cosmic diurnal cycle and thus developed into the social attitude of day-work and night-rest of most of mankind.

It has been shown in many studies of human efficiency that, depending on the activity, fatigue or boredom will bring about a measurable performance decrement in the relatively short periods of a few hours. The mid-morning and mid-afternoon drop in efficiency brought about the "coffee break" system. Due to the diurnal cycle and the consequent social behavioral pattern, most people would go to sleep during this period of efficiency drop. Some lucky ones, who can afford it, will take a short nap anyhow in the afternoon.

Not only the wakeful state has these fluctuations but, as various studies have demonstrated, also sleep shows periodic phenomena. It should be mentioned here that this problem is even more complex due to the many other parameters which influence sleep and wakefulness.

In astronautics where the two major factors, namely, the diurnal cycle and the social work pattern are not present, it will be much more advantageous to fall into a cycle different from the eight hours to sixteen hours rhythm.

It has been suggested that for prolonged trans-lunar missions a four hours wakefulness, four hours sleep cycle should be adopted. Such a cycle seems proper for the work profile proposed for certain space missions, the details of which are beyond the scope of this paper. The diversity of the duty cycle at various types of missions will, however, require an even more flexible pattern where the astronauts may need varying sleep-wakefulness periods to fit the pattern of the mission. This proposition, however, will meet with difficulties due to the strong habituation of the eight-sixteen hours rhythm.

There is a considerable amount of literature available on the physiology and biophysics of sleep, but relatively little is known on the psychophysical factors of artificially induced sleep. Yet the problem of sleep is quite important in the human factors of astronautics because of the need for maintenance of high efficiency of the man-machine system.

Sleep is the best long-range remedy for all variants of fatigue which must be reduced to a minimum to avoid performance decrement, for which certain space tasks are quite intolerant.

Besides this recovery from fatigue, sleep may be used for sleep learning or reenforcement of learned information. This latter aspect will become significant in aiding recall of little used data which, however, may be needed suddenly without permitting the necessary time for consulting books or tapes.

II. BIOLOGY OF SLEEP

Although sleep is a most basic bodily function, amazingly little is known of its fundamental mechanism. This, in spite of the virtually thousands of scientific papers dealing with the subject.

Therefore, a brief review is given here preeenting some of the more important information

a) Consciousness vs. sleep

Even the terminology of consciousness and sleep is quite confusing. There is no space here, of course, to discuss the semantic subtleties of the difference between consciousness and wakefulness, etc. In rough approximation it can be stated, however, that during consciousness the organism is meaningfully perceiving environmental stimuli and is capable of decision making and able to activate psychomotor functions.

In general there are three essential aspects of sleep:

1) the quality or depth of sleep
2) the quantity or duration of sleep
3) the rhythm or periodicity of sleep

Maybe the greatest difficulty in sleep research is the measurement of the depth of sleep. A quantitative evaluation of this factor is most pertinent because of the wide spectrum between drowsiness and coma. The physiological and psychological aspects of sleep depend greatly on the state of sleep.

There are a number of measurements proposed for the determination of depth of sleep.

One system uses the numbers 1 to 4 to indicate states of sleep, another uses the letters A to G with intermediates denoted by a + or - sign. Again others refer to slow, rapid or no eye movements which are associated with states of sleep. Fair quantitative information can be provided with good reproducibility, using electroencephalograms, which will be discussed later.

Furthermore, a number of stimulus gradients were applied for depth-of-sleep curves. Auditory, visual, tactile and pain sensations were used with little reliability.

Stupor and hibernation, although these stages can be considered as extremes of sleep, will not be discussed in this paper.

The duration of sleep is strongly dependent on the age of the person. In adults an average of 7.5 hours sleep per 24 hours seems to be a basic human trait because it holds for all races of Homo sapiens, at least for those who live in the temperate zone.

The sleep-wakefulness periodicity is anchored very firmly in the human physiology. Body temperature changes and other physiological collaterales of sleep do not change readily even after prolonged training. In a test, after many weeks of training, subjects of 12 hours sleep cycle reported no overt discomfort but their body temperature rhythm remained corresponding to the former 8-16 hours cycle. Visceral activities also seem to resist modification of sleep cycles.

Meteorological factors, which have such a profound influence on sleep, are meaningless for astronautics due to the controlled environment of the closed ecological system of the crew compartment. Cosmic cycles may still have bearing on sleep in the space crew compartment but research data on this are incomplete.

b) Anatomy of Sleep

Among many investigators who have contributed to the anatomy of sleep, M. Jouvet is outstanding. According to Jouvet two different neurophysiological systems are controlling sleep. These are:

1) the telencephalic activity, which has inhibitory effect on the ascending reticular system and originates in the cortex of the cerebral hemispheres,
2) the rhombencephalic phase, which has an inhibitory connection to the limbic system in the midbrain and the reticular cells in the pons. These reticular cells are responsible for the maintenance of alert wakefulness if stimulated from peripheral sense organs.

Of course, since Aristotle, many thinkers and physicians and other related professionals have developed various theories of sleep, none of which were satisfactory, so far.

Some of these theories are relating sleep to the circulation in the brain; others to temporary synaptic disconnects due to amoeboid motion of neuroglia cells. Again, others are advocating that there is an undetected sleep center in the brain which is responsible for the loss of consciousness. The theories of the accumulation of a specific metabolite called "sleep poison" or hypnotoxin as well as the instinct theory of sleep were explored with insufficient results.

The fact of the matter is that the mechanism of sleep is not yet well understood.

c) Physiology of Sleep. Sleep feeding relations.

As indicated before, sleep has a profound influence on many physiological parameters. They are so numerous that only a very few can be mentioned here.

The pH and the blood sugar content of the blood do not change significantly during sleep. There is a slight change in the calcium equilibrium in the body fluids during sleep, but no other noteworthy alteration has been observed during sleep. The pulse-rate, as expected, decreases in the sleeping persons and as an extreme a rate of 45 has been observed. The blood pressure in the body also decreases with drops of 20 to 30 mm Hg, depending on pressure in wakefulness. These figures may account for the anemia in the brain during sleep.

What is interesting is that epinephrin, which causes a strong increase in blood pressure in the wakeful person, fails to do so during sleep.

The symptom of snoring is not considered a pathological condition and does not influence sleep except for those persons who have to share the environs of the snorer. Respiration is altered by sleep but the reports on the details are quite conflicting.

Digestion in its biochemical phase is not much influenced by sleep. Some glandular change has been reported, such as the accumulation of bromine in the anterior lobe of the pituitary gland. The change in body temperature is not considered to be related directly to sleep but to the lack of muscular activity.

d) Psychology of Sleep. Dreaming.

Considerable information is now available concerning dreaming. The results are quite far reaching and show that dreaming is a necessary physiological-psychological activity.

During the dreaming phase the blood pressure is generally higher and the variability greater than during other phases of sleep. It is considered, however, that the blood pressure might not be as specific a concommitant of sleep as other factors, rather that it might be due to emotional changes during dreaming.

In a spacecraft, where extensive physical exercise is unlikely, the maintenance of sleep state which is accompanied by rapid eye-movements and which contains the subjective experience of dreaming, is an important factor of life support. It should be mentioned, however, that the correlation of ocular motility in sleep with dreaming is not precisely established and further research is indicated.

The functional integrity of the astronauts is related also to sleep control from a psychiatric point of view. It seems that dreams and hallucinations are brought about by a similar mechanism; thus, psychosis and dreams are related mental processess. From this point of view it is essential to monitor carefully the sleep pattern of the astronauts because sleep disturbances are characteristic for the onset of functional psychosis. It should be pointed out, however, that the EEG and the eye-movement pattern are similar in the normal and the schizophrenic persons; thus, such telemetered information would not give significant indication of schizophrenia occuring in the astronaut.

It was found that dreaming is associated with rapid eye-movements and the corresponding EEG is of low voltage, rapid and without sleep spindles.

Between the dream periods are distinct intervals of cyclic pattern. At these times there are no eye movements and the EEG is entirely different by showing the so-called sleep spindles and about 1 cps delta waves. However, cyclic changes corresponding to dream experiences are not restricted to symptoms occurring in the EEG, but significant changes are manifest in muscle tone, frequency and depth of respiration, skin resistance, blood pressure, etc. These symptoms are easily measured and lend themselves well for telemetry.

The rapid eye movements are not continuous during dreaming but the 4-5 dream experiences per night are normally uninterrupted for longer or shorter times depending on the person's age. For the age limit of 20 to 30 years the total time of dreaming during 7-8 hours of sleep, is one hour and 30 minutes, with amazingly little individual variation, and occurs about every 90 minutes.

For the mental hygiene of the astronauts it seems important that they should have good wholesome dreams every 5-6 hours to avoid "day dreams" or better called "hallucinations." According to theoreticians in this field, dreams are instinctual drive discharge processes in the id, the significance of which has been demonstrated by dream deprivation experiments. They lead to hallucinations. Obviously, the dream-sleep cycles are governed by quite deep-seated physiological, innate mechanisms. Freud called dreams "safety valves." If these valves are closed the pressure for discharge mounts until the "dreams" occur during wakeful periods. The process of dreaming is called by Jouvet "rombencephalic phase," which seems to be a somewhat cumbersome nomenclature, but has anatomic significance.

It is believed that dreaming is also connected with neuro-humoral secretion. Thus, there might be developed in the future a biochemistry of dreaming but it is likely that the understanding of dreams will need purely psychological interpretation.

There are certain similarities between dreaming and the effects of sensory deprivation. This seems to indicate that dreaming is in fact independent from sleep but is related to the relative absence of environmental stimuli which are essential for wakefulness and the various aspects of adaptive behavior.

Although we cannot enter here into the detailed psychodynamics of sleep, it seems that for the proper functioning of astronauts on long missions we will have to control not only the sleep cycle, but will have to manipulate also the quality and the duration of dreams.

e) Sleep Deprivation

Chronic disturbances in indoctrinated sleep cycles may lead to anomalies of waking experience and behavior. It can easily lead to psychosis.

If muscular activities are excluded it is impossible to deprive a normal person of sleep for longer than two days.

Prolonged sleeplessness causes irritability, irrationality and hallucinations. It causes also marked and increasing performance decrement.

There is a stage or state of mind where the concept of wakefulness becomes transitory to sleep.

f) Psychophysics of Sleep. EEG.

If one is looking at a flickering light, rhythmic signals appear on the EEG which change with the flicker frequency. This indicates that those signals were generated in the visual projection area of the brain.

It is significant that the flicker is generally accompanied by sensation of motion and patterns.

The action potential which is concommitant with every biophysical or dynamic activity of every cell is manifest also in brain function.

The best understood pattern of electroencephalograms is that of sleep. It enables us to clearly distinguish between the various stages or levels of sleep and arousal.

The various typical patterns of the EEG are designated mostly by the Greek letters alpha, bets, gamma, delta and epsilon. The alpha pattern, which appear also by slight drowsiness, is 9-12 cps with about 60 microvolt amplitude. Beta rhythm is of low voltage, where the alpha pattern is replaced by small undulations and indicates definite sleep. Gamma rhythm is characterized by the so-called sleep spindles and wave trains of 14-15 cps with 20-40 microvolt amplitudes. These are superimposed on irregular slower waves. Delta is the designation of high amplitude (300 microvolt) very slow 1 cps waves with irregular sleep spindles. The Epsilon stage shows random pattern with even slower Delta waves of 5-20 sec intervals. This stage is characterized also by the complete absence of sleep spindles.

The total picture is, however, not as simple as indicated here and large amounts of literature are available for the interpretation of further and complex details.

III. SLEEP CONTROL

The human mechanism has an apparent shortcoming as compared with a bionic device, namely that it cannot be switched "on" or "off" as needed. Of course, there are bodily activities such as recovery from fatigue, etc. which require the proper functioning of the body during a non-conscious period. This occurs, however, at a lower metabolic rate, although it can be said that during dreaming sleep we are more intensely alive than in the conscious or waking existence.

But there are a number of ways which permit a certain limited control of sleep. They are briefly mentioned below.

a) Training and Indoctrination

It is possible that by prolonged training most individuals will acquire a four-four hours rhythm. It must be expected, however, that such terrestrial training may initiate neuroses, because the forced rhythmic behavior may not coincide with the long established physiological "clock" mechanism. This condition will be further aggravated by the weak environmental signals which will reach the person even in his confinement and which tend to reenforce the deeply embedded original pattern. The astronaut will be especially susceptible to these weak signals as a consequence of the stress of sensory deprivation caused by the confinement in the space compartment. Of course, the simulation of certain nocturnal noise, to which the urban population is acclimated, may be used to override the weak diurnal noises. However, this method may not be sufficient to provide sleep.

b) Drugs

The use of hypnotics must be discarded because our present inventory does not contain a compound or mixture which would initiate sleep rapidly nor does the compound have a well defined four hours of duration and permit a quick arousal without any after-effects.

There are two distinct types of drugs, the hypnotica which facilitate sleep and narcotica, which cause in many people irresistible and deep stupor. However, hypnotica, if administered in larger quantities will cause a state called narcosis, or, if the product is applied through respiration, anesthesia.

c) Hypnosis. Post-hypnotic Suggestion.

This method cannot be recommended because of the limited duration of its effect. It is thinkable that a combined post-hypnotic and autosuggestion technique which would utilize an artificial clue, such as a tape recording, may work for a longer time but this method has not been demonstrated to be reliable enough.

Beyond the aforementioned practical space applications, it is significant that the below mentioned technique of electronically induced sleep could contribute to our understanding of the various hypnotic states. From a few experiments it seems also that this technique will have therapeutic values inasmuch as those neurotic patients who resist hypnotherapy could be brought into a state of consciousness where their excitation and hostility is reduced enough to make them responsive to hypnotism.

d) Visual and Audio Stimuli

It is a well known fact that certain rhythmic photic stimuli or monotonous audio signals have the tendency of hypnotic sleep induction.

The effects of flickering light has been under study for about 200 years and it has brought us quite close to being able to disrupt consciousness and thus manipulate the onset of sleep. Previous reports have indicated that the limit of the results was a slight hypnotic effect with drowsiness and occasionally with unpleasant side effects. The impairment of cognition by flickering light has also been described.

e) Electrical Current

Japanese and Russian reports, together with the research of the present writer, which he has performed in the past years, suggest that the solution to the problem of sleep control of the astronauts may lie in the area of cyclic electronic stimulation of a so far unknown resonance area in the brain, which reduces cognition and willful activities.

It is known from research here and abroad that electrical stimuli of a certain frequency, if applied to the brain, will cause drowsiness or sleep.

The electrical influence of consciousness can be subdivided into three distinctly different types:

1) electroshock
2) electronarcosis
3) electrosleep

Electroshock was used extensively in certain mental diseases such as schizophrenia and sometimes to supplement prefrontal lobotomy. Lately, it is applied less frequently. The effect of the relatively high current density causing artificially a seizure, with temporary blocking of respiration.

This type is applied primarily only to psychotic patients and does not concern us here.

Electronarcosis, also called electroanesthesia, is a modified, subseizure electronically induced loss of consciousness, without drastic effect on respiration. However, convulsions are frequent and the application of muscle relaxants is indicated. Sometimes the extreme rise of blood pressure puts this technique into expert hands. In spite of this, electronarcosis has definitely a place an astronautics.

Although the character traits of astronauts are as carefully determined as possible, under many months or years of lasting space-stress some psychosis may occur. Since the reaction to deprivation, of which there will be many in space, is the defense of a break with reality, psychotic persons will be greatly prone to hallucinations and delusions.

Many reports indicate that electronarcosis can be successfully applied to alleviate such conditions.

Since electronarcosis has a hypnotic effect to the reflexless stage it can easier be applied in case of medical emergency than inhalation anesthesia or drugs which could be in short supply in space, but electricity will be always available.

Over 60 years ago electronarcosis was successfully applied by Leduc at the beginning of the century, on rabbits.

The applied frequencies vary from 60 to 700 cps and the current is reported between 5 and 250 milliamps. Thousands of narcoses were administered, to men and animals, in this fashion.

Good results were obtained with 700 cps sine wave signals with 12-50 volts and 50 to 100 milliamps. The electrodes were placed on the temple. The equipment consists of an oscillator, an inverter and a push-pull amplifier.

Electrosleep produces a shallower state of sleep, which persists for some time after the current is cut off.

There is some uncertainty whether by applying such weak signals sleep is due to the current or suggestion. Whatever the underlying mechanism may be, the hypnotic effect of 9-14 cps sine wave signals on many test persons has been established.

IV. ELECTRONIC TECHNIQUES

a) Applications Published

In a critical review of the available literature it became evident that electrical signals fed into the brain through skin electrodes, have a most confused pathway due to the very noisy electrical environment. This situation is amplified by the characteristics of the low frequencies used for stimulation which have a diffused propagation under the dielectric of the brain.

Therefore, it occurred to the present writer that the signals should be channeled through the established pathways of afferent neurons. Here evolutionary and homeostatic mechanisms provide a noise-free input.

Japanese and Russian reports, together with the research of the present writer, which he has performed in the past years, indicate that the solution of the problem of sleep control of the astronauts lies in this area.

The writer has investigated this problem and brought about preliminary results which have demonstrated the feasibility of his approach. The applied equipment and a clinical and statistical evaluation, however, need further research and development. The hypothesis was that if one can generate a rhythmic stimulation of the optic nerve, the frequency of which coincides with the alpha rhythm of the person's EEG, then by presently indefinable resonance the "sleep center," or the rhomboencephalic reticular fibers will adopt the periodicity and as a consequence sleep will occur, provided the noise in the brain is below a certain arousal value. Research has been performed mainly on animals. Some of the information available on humans is mentioned here.

Japan's National Railway Labor Medical Laboratory reports that 10 cps and 100 microamperes applied to insomniacs eye-lids will put them to sleep within seconds.

Gilyarovsky in the USSR applies a pulsing dc current in the frequency band of 1 - 20 cps, with square wave pulses of 0.2 m sec duration.

Vransky in Bulgaria applied 0.5 - 2.5 volt and 0.2 - 1.5 mamp in form of ac current.

In the USA research is in progress in New York, Chicago and other places, with a broad spectrum of experiments.

b) Experiments of Present Writer.

In the exploratory experiments, of which this is a brief report, 52 persons were used, 18 females and 34 males, who were in the age bracket of 17-22 years. A few of the experiments were repeated with the same person on several days. The persons were subjected to the experiment after receiving explanation of the purpose and technique. In cases where the subject showed anxiety, which was frequent, the electrodes were applied and the effect was demonstrated. The electrodes were then removed and the subject was left alone on the reclined chair while the experimenter busied himself with other work. Within five to ten minutes the subjects themselves requested continuation of the experiment. From then on the experimenter did not communicate with the subject to avoid hypnotic suggestion.

In many cases the subjects selected the "most pleasant" frequency and the "best" current by themselves adjusting the signal generator. This activity must have preconditioned the subject because in these cases the hypnotic effect seemed to occur sooned than in cases where the frequency was pre-set.

Based on the declared results of the advertised "heart-beat comforter" for babies, it was considered to explore whether a monotonous audio stimulus is synergistic to the effect of photic stimulation. Therefore, the heart sound was chosen and a tape recording was made from an audiocardiograph. The tape was edited to remove the sound of arhythmias and the peristaltic noise. Without exception the subjects considered the sound of the recording as distractive and it interferred with the effects of photic stimulation. It seems possible, however, that other less complex audio signals than the heart sound could be effectively used if applied at the low frequencies of 1 - 2 cps. This is indicated by the experiment by which a small relay or solenoid was strapped on the vertebral column of a rabbit in the sacroiliac region and made to tap the vertebra at set frequencies. The results were inconclusive because the problem could not be explored with statistically satisfactory data in the available time. In some cases the animal went to sleep; in others it acquired catatonic posture or showed almost epileptoid behavior. In other instances when the animal was stimulated at other parts of the backbone there seemed to be no effect. In these experiments it is uncertain, of course, if the stimulus is primarily tactile or audio.

The use of weak audio signals which are pleasurably but subconsciously associated with the person's individual environment at the time of going to sleep, has been considered.

The photic stimulus was generated by a signal generator with variable frequency. The frequency was selected by the subjects who were permitted to scan between 1 cps up to the frequency of flicker fusion. All subjects tested so far have chosen a frequency between 10 and 14 cps. It is thinkable that the hedonic value of their choice in frequency may have significance in some aspects of hypnotherapy.

The electrodes were made in the shape of eyeglasses with the glass replaced by a metal screen which was covered with a thick pad of gauze. When used, the pads were moistened with saline solution. To the ends of the temple of the eyeglass frame was fastened a rubber band which assured proper fit and contact to the eyelids. The connection from the electrodes to the signal generator was made with alligator clamps on a shielded cable.

The input signal was sinusoidal and was applied across the eyeballs. Depending on the skin resistance, the current was 20 to 50 ma. It is thought that such a low frequency current probably travels within the sclera to the optic nerve. This is fortunate for the protection of the retina at prolonged exposure. Higher currents in the 200-300 ma range may propagate partially in the choroid, the effect of which must be ophtalmologically tested. In all experiments performed not one subject showed even the slightest temporary aftereffects in spite of several repetitions. Presumptive data indicates a broad margin of safety for this technique.

The observation of the depth of sleep by the eye movement was not possible because the electrodes covered the eyes. Therefore, the onset of sleep was measured in many cases by a light touch of the underarm hairs with a camel-hair brush and instructing the subject to report with a slight motion of the index finger. The tactile stimulus was applied at random intervals of about once a minute until the report failed. Sleep occurred in most of the cases within five to eight minutes. By repeating the experiment on different days with the same person there was at first a decrease in the time, but by further repetition an increase was observed in the time required for the effect.

The room was a quiet environment (10-15 db) but was not fully soundproofed to avoid possible effects of sensory deprivation.

From these preliminary experiments it seems to be evident that a sinusodial wave pattern in the low frequencies applied as photic stimuli produce in most cases a hypnotic state. The frequency range was 10-14 cps; thus, the effect is not identical with electronarcosis, which operates at much higher frequency and higher current.

The mechanism of the onset of sleep seems to respond better to the sine wave pattern than to square waves or to signals with extremely short duration, as was used by Russian investigators.

It seems also that the conscious level of the subject was about A- on the Simon-Emmons scale, although the favored frequency of about 14 cps corresponds to the sleep level C on the same scale, which indicates the appearance of sleep spindles.

The plans call for further experiments with larger numbers of subjects, better controlled conditions and EEG, as well as biochemical and physiological monitoring, such as eosinophil count, lactic acid production, etc. The spectrum of frequency will be also extended to five seconds or even fifteen seconds per cycle to explore a possible hypothalamic resonance effect in the delta activity range.

It is expected that, from the presently available laboratory equipment, a prototype will be developed which can be applied to sleep control in the space crew compartment.

Such a device must be:

1) applicable and effective to the majority of astronauts
2) proven harmless
3) free of side effects
4) highly reliable and maintainable
5) insensitive to acceleration
6) insensitive to nuclear radiation
7) light weight and small

Good progress has been made in the R&D of such a design.

Acknowledgment: I wish to acknowledge the conscientious efforts and achievements of my student assistant, Mr. Ernest Lado, Jr., who diligently helped me in these experiments.

V. CONCLUSION AND FUTURE DEVELOPMENTS

Man spends about one-third of his lifetime sleeping. This in itself indicates the significance of this state for the physiological and psychological homeostasis of a normal person.

Unfortunately, the sleep-pattern of man does not match always the requirement of our modern and sophisticated socio-technocratic life. Therefore, recently it became necessary to study the mechanism of sleep. The aim of such investigation is, besides the scientific information, the desire to manipulate sleep.

What is known is, in essence, the fact that monotonous repetition of audio or visual stimuli has frequently induced sleep in a number of subjects. This effect has a number of dependent variables, such as frequency spectrum, environmental factors and variables of certain mental states.

In addition to the condition of insomnia, this problem received impetus by the design of the work profile of long-range space missions. Here the rest-work cycle is radically different from the terrestrial nocturnal-diurnal human pattern. Because of the - some times - highly taxing performance requirements it is imperative that the astronaut should fall into restful sleep at any regular or irregular intervals. Of course, drugs which cause drowsiness or other side effects are out of the question.

It has been confirmed by the writer that sinusoidal electrical stimuli of the optic nerve in the 9 to 12 cps range will bring about sleep in most subjects within minutes. The current applied is below the sensory threshold of the skin and the homeostatic effect is fair provided arousal threshold is not exceeded. In other words, the effect shows stability against overriding stimuli. The created flicker seems to dominate, if applied long enough, even over otherwise strong arousing audio stimuli such as soft jazz music.

The biophysical or psychophysical problem of sleep control has a broad spectrum for experimentation. Large numbers of individual experiments have to be carried out for statistical evaluation of the results. They will demonstrate the considerations which are needed if one deals with a single factor of a complex system.

It does not seem far fetched to expect that within a relatively short time man will develop equipment which may be pre-set to alter the biological clock of sleep.

The time might even come when implanted bionic will control sleep and wakefulness. They could be governed by biophysical or biochemical symptoms of fatigue; thus would eliminate the over-taxing of our physiology and prevent the exposing of our psyche to undue stress. Such a condition will significantly increase the life expectancy of mankind. We could live 150 or 200 years, taking even into consideration the lack of regeneration of our cardiac tissue.

Of course, when such time comes, our activity pattern will be greatly changed. You may feel a prickle on your skin above the implanted bionic device. This could signify: "You are working too hard, better go to bed because in five minutes I will make you sleep." On the other hand, after two hours it might wake you up and if you choose to stay in bed it will sense the reduced rate of respiration and circulation and will deliver continuous mild electric shocks which signify: "You are lazy. I know you are well rested. Get up and do whatever you want to do."

And this, ladies and gentlemen, is not any more science fiction. It seems that a good old standby, the alarm clock, is going out of fashion.

Figure. 1

SELECTION AND EVALUATION
OF AN
EXTRA-VEHICULAR LIFE SUPPORT SYSTEM CONCEPT

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SUMMARY

Of the multitude of obstacles that confront the scientists and engineers engaged in making Project Apollo a success, those directly associated with life support appear the least defined. They are not defined clearly because they combine physiological and psychological factors peculiar to the human machine with the parameters upon which heat and mass transfer, mechanics and kinetics depend. Because Project Apollo is a manned mission it follows that all the benefits which can accrue due to man's participation must be realized. . . . . . that he can not be hampered from accomplishing his tasks, be they thinking or doing, because of the limitations imposed by his portable environment. It is the intent of this paper to discuss some of the interesting areas that have been uncovered to date in the optimization of a Portable Life Support System.

The system is intended for extra-vehicular use - either in deep space or on the surface of the moon. It consists of a full pressure suit assembly (which includes an overgarment designed to act as a thermal barrier) and the back pack which contains all the equipment necessary for replenishing and conditioning the internal atmosphere. Provisions are also included for waste management, communication, performance monitoring and telemetry of performance and physiological data, etc.

At the onset of the Apollo Space Suit Development Program intensive theoretical heat and mass transfer studies were made to establish a procedure through which the performance of the ventilation system could be predicted and/or evaluated. The mechanism through which a gas stream removes heat from the suit loop is very complex. The pressure suit does not provide an adiabatic wall, so the ventilation stream must be capable of absorbing the amount of heat which flows in from the surroundings.

Because it is desirable from a weight standpoint to select a low ventilation flow rate (12-16 cfm), the contact time between it and the body can be expected to be relatively long. Moreover, except for extraordinary heat leakage conditions, the highest temperature in the suit portion of the loop is that of the body surface. Due to the long contact time and the nature of the return flow path, the man-suit system is a rather effective heat exchanger. Thus, exit flow from the suit is close to skin temperature. Hence, for a given inlet temperature and weight flow, the sensible stream heat capacity is almost constant. If the heat leakage from the environment is greater than this capacity, the subject adsorbs the excess and returns it to the stream in the form of latent heat. If, on the other hand, the heat leakage is negligible or negative (leaving the suit), the entire sensible capacity of the stream is used to cool the subject and only the excess of metabolic heat over the sum of sensible stream capacity and heat leak (outward) is transferred latently.

Because system power requirements and, therefore, weight depend greatly on the ventilation flow rate, it is imperative to select the minimum flow rate that will maintain the subject in thermal equilibrium at the anticipated metabolic activity rate commensurate with safe and comfortable physiological tolerance to relative humidity and temperature. Thermal control is maintained by a water boiler. Humidity control at the suit inlet is accomplished by maintaining the suit inlet flow saturated at water boiler exit temperature. Excess water is removed by a water separator.

The weight of each of the components of the PLSS as a function of total system heat rate, total integrated mission heat load, local volume flow and other independent variables can be described by an equation. Thus, an estimate of total system weight can be made as a function of total metabolic activity. For a given integrated load, the shape of the mission activity-time profile also affects system weight. The weight of material required for CO_2 removal and the amount of water stored for cooling depends on the length of time during which peak loads are maintained and on how close to the end of the mission they occur.

The results of a parametric study in which latent heat, system weight, volume flow, and total system heat load are plotted as functions of total metabolic load and heat leakage have been presented; the relationships linking PLSS weight, system electrical power, system pressure drop, and metabolic load have also been included.

The testing of the prototype PLSS has demonstrated that the equipment can, indeed, perform the mission for which it has been designed. Moreover, when an instrumented unit was committed to manned test, the performance of the SSA was satisfactory. More important, the data obtained to date will be an invaluable tool in the design of the next generation of Portable Life Support equipment.

SELECTION OF A DESIGN POINT

Before any PLSS concepts could be considered, it was necessary to estimate the rate of metabolic energy expenditure of an astronaut during a typical Apollo mission. This was necessary because the major portion of the heat that must be dissipated in such a PLSS comes from that produced by the occupant's metabolism. At the onset of the program, a ground rule was established by NASA which dictated that the system should be capable of handling an average metabolic load of 11,300 BTU/day. Therefore, a system designed to dissipate a metabolic load of 500 BTU/hr was proposed. In a short time after receipt of the contract to develop the PLSS system, it became apparent that the metabolic load requirement of 500 BTU/hr was unrealistic. At that time a theoretical study was made in an attempt to assess the effect of the lunar terrain, the reduction in gravity, the encumbrance caused by the suit, and similar unknowns on the metabolic energy expenditure rate of the astronaut. It was also necessary at that time to make a first cut, however premature, of a mission task analysis. Thus, activities such as walking, setting up of photographic and astronomy equipment, gathering of geological specimens, and similar activities that probably would be performed during such a mission were assigned nominal metabolic costs. These rates came largely from available literature which has been generated at several universities and biomedical laboratories. For example, the metabolic rate associated with walking at a speed of 3 miles/hour was taken to be somewhere between 1200 and 1800 BTU/hr. This value is considerably affected by the slope over which the subject is walking and by the amount of weight he is carrying. It is also affected by the mechanical efficiency with which he is performing that task. For the purpose of this study it was assumed that the mechanical efficiency of the human body during a walking maneuver is zero; that all the energy consumed during walking is converted to heat. It was, therefore, necessary to design a system to be capable of dissipating the entire metabolic heat generated by the body which is associated with a given task.

The increase in potential energy that is achieved during walking up a hill or in lifting an object from one height to another which can be considered as a useful work output was, for the purpose of this investigation, assumed to be removed by the system as heat load. The underlying reason for such an approach is that the pressurized suit requires some amount of work just to move it which does not escape from the system as useful work. This is defined as suit encumbrance. This encumbrance was later defined during the test program. The encumbrance caused by unknown lunar terrain conditions could only be estimated. Some motion studies were made of men walking at various speeds to determine the rate at which the center of gravity of the body moved up and down and the amplitude of such movements. This information could then be applied to a man of given weight walking at a given speed to determine the associated work. Based on these studies it was decided that for a four-hour typical Apollo mission the average metabolic energy expenditure rate for the entire four hours is 930 BTU/hr. This number includes rest periods. The peak metabolic rate was estimated to be 1600 BTU/hr.

DETERMINATION OF EXTERNAL HEAT LEAK

In addition to the heat generated within the man-suit system, it was found, based on estimates of lunar surface temperature, that a considerable amount of heat would leak into the system unless some type of thermal insulation was utilized. The design point selected was the subsolar position. At this point it was assumed that for one reason or another the astronaut would be in a horizontal position such that the solar flux was exposed to one-half of his body. The lunar surface temperature at the subsolar point was taken at 250°F. At this combination of heat inputs the use of a super insulator is mandatory.

From an investigation of several super insulator materials it was found that the heat leak into the suit could probably be held to under 150 BTU/hr. This is only accomplished through the use of an outer surface for which the emissivity at infrared wave lengths is rather high and the ratio of absorptivity to solar wave lengths over emissivity was low. An equilibrium surface temperature of 175°F was calculated. This was well below the upper limit to which this material could be subjected.

GENERAL PLSS REQUIREMENTS

In order to compare various systems, it is desirable to state and summarize the specific tasks that must be accomplished by the back pack. The regulation and supply of oxygen at a pressure level of 3.5 psia was selected. For a subject working at 1600 BTU/hr the average oxygen consumption rate is 0.26 lbs/hr. For the average metabolic rate of 930 BTU/hr the metabolic O_2 consumption is .18 lbs/hr. Therefore, the oxygen supply must be sized to provide this 0.18 lbs/hr for the entire mission in addition to the quantity of oxygen which leaks from the system. Moreover, in a design of oxygen pressure bottles there is an ullage factor which represents that quantity of oxygen which cannot be removed effectively from the bottle. The bottle pressure that was selected as a ground rule was 850 psi because this was the pressure of the oxygen that will be stored on the Apollo Command Module and LEM vehicles. This would, therefore, enable recharging of the bottles to be accomplished without too much difficulty. The control of contaminants within the system is another objective of the PLSS. The principle contaminant, of course, is the carbon dioxide generated by the subject during the performance of his tasks. For a man working at the average 930 BTU/hr the production rate of CO_2 is in the order of 0.182 lbs/hr.

Thus over a four-hour mission components had to be designed to absorb approximately 3/4 of a pound of carbon dioxide. It must do this while not allowing the partial pressure of inspired CO_2 to exceed the safe tolerable limit. For periods in the order of four-hours, this maximum limit was considered to be one percent of a sea level atmosphere or 7.6 mm Hg. The use of LiOH and LI_2O as CO_2 absorbers was evaluated. Because reliable data on LI_2O was not available and because LiOH has already been used successfully in this type of application, it was selected as the absorber material. The back pack must also provide a method of recirculating the oxygen within the loop. This is a most important function. The back pack may incorporate the ultimate in heat rejection equipment, but if the heat (both latent and sensible) is not transported from the suit to the back pack the equipment is useless. Therefore, the use of a circulating fluid is necessary to perform the heat and mass transport function. Oxygen is the obvious choice for the transport fluid because its presence is already required for metabolic needs. The recirculation flow is also important from a standpoint of helmet carbon dioxide purging. Just as in the case of a heat exchanger being ineffective without a transfer medium, the carbon dioxide removal canister would be completely useless if the carbon dioxide were not transported from the oro-nasal area to that area of the back pack which removes CO_2 from the stream.

The function of thermal control involves humidity control and temperature control. Temperature control is not merely reducing the temperature of the stream to remove the sensible metabolic load. It also implies the dissipation of all other heats created in the system. For example, that produced during the chemical process in which CO_2 is combined with LiOH, or the heat that is produced by the work put into the stream by the fan, etc. Humidity control is required because the driving force available for transfer of latent load is a function of the gradient in vapor partial pressure.

Most of the heat generated by the subject is given off in the form of latent heat. This is so because the flow that is circulating within the system has been selected to be the minimum rate commensurate with physiological well being of the subject. In order to increase the portion of metabolic heat which is removed sensibly, higher coolant flow rate is required. This implies a larger fan and power requirement which rapidly increases the weight associated with energy storage (battery or pressurized O_2). This minimum rate is strongly influenced by the CO_2 purge requirement. Therefore, it is observed that one of the fundamental ground rules that was applied in the design of this system was that it was necessary to maintain the subject in a sufficiently good state of comfort to perform his intended tasks. It was not, however, possible to make him "comfortable" by everyday standards because this is too great a luxury. The maintenance of 70 percent relative humidity in the center of the comfort regime in a psychometric chart requires a system volume flow which could only be obtained with a huge weight penalty. The back pack must be equipped to dissipate the latent load at a level such that the partial pressure of water vapor into the suit gives rise to a partial pressure gradient between skin and stream which provides transfer of sweat at the same rate at which it is being generated. When the latent load is removed in the back pack, a considerable amount of the subject's sweat which was in the form of water vapor is condensed to liquid water. This, therefore, necessitates the use of a water separator. If a water separator were not used the liquid would simply be re-entrained into the stream. The stream would then enter the suit at a high value of dew point which would prevent mass transfer from the man to the coolant.

Another function of the back pack is that of providing a prime mover. The recirculating stream can be powered by several means but the energy to do this must come from somewhere. Several approaches to this requirement are discussed in the following section.

A very important function of the back pack is that of communication and telemetry. It is desirable, if not necessary, to be able to monitor the physiological well being of the astronaut as well as the functional performance of the back pack at all times. Therefore, a telemetry and communications system must be incorporated into the back pack. Typical physiological parameters that can be monitored are heart beat, respiration rate, and body temperature. Within the back pack continuous measurement and telemetry of suit inlet temperature, battery voltage, oxygen bottle pressure and recirculating system pressure are envisioned. Finally, the back pack should contain any warning devices and emergency systems that are considered to be necessary either for improving the reliability of the system or the ability of the astronaut to assess any problems that may come up during operation.

ALTERNATIVE PLSS CONCEPTS

Although there are many combinations of components that would satisfy the objectives outlined above, the discussion of alternative system concepts will be limited to the ones which appear to be most practicable.

Figure 1 shows schematically a system in which the recirculating flow is pumped by a fan which in turn is powered by a battery driven motor. The LiOH canister could be located before or after the fan. An optimization study based on weight was conducted to determine which of these locations was most desirable. It was found that the penalties that accrued from locating the LiOH canister downstream of the fan in terms of decreased canister performance more than offset the advantage (lower power requirement) that resulted from putting the fan upstream. Therefore, the LiOH canister was placed upstream of the fan.

The water boiler was located as close to the suit as possible so that a low inlet temperature and dew point could be realized. The line from the water separator to the water boiler returns condensed sweat to the water side of the water boiler. This is a unique concept in as much as it makes use of the sweat twice; the first time being when the man gives off the latent load. When the condensed sweat is boiled off as steam in the water boiler, it is being used to cool the recirculating stream. The use of sweat in this fashion enables a more economical sizing of the water boiler storage sump. Any water that could be obtained as a result of sweating does not have to be stored at the beginning of a mission.

Figure 2 is a schematic diagram of a system powered by an ejector. Stored oxygen is used as the primary flow in the ejector. The oxygen can be stored in either gaseous form at the previously stipulated 850 psia or, if practicable, at any higher pressure consistent with tolerable stress values in the bottle. Cryogenic oxygen storage has also been considered as a possibility in this type of application. However, for the small quantities of oxygen that would be necessary in a four-hour mission of this type, it was found that with the weight of insulation required to maintain the low temperature of the cryogenic oxygen, there was no appreciable savings in bottle weight. The overall weight penalty associated with cryogenic storage is approximately one pound of tank per pound of oxygen stored. While this compares favorably with the over two pounds of tank per pound of gaseous O_2 stored at 850 psia, the additional quantity of oxygen required in an ejector system more than accounts for this difference in weight. Ejectors have been tested which exhibit primary to secondary flow ratios of 7 to 1.

It is possible that values up to 9 to 1 can be realized, but it is questionable as to whether or not the required system pressure drop can also be made up when a ratio of 9 to 1 is utilized. Note that in such a system, the primary flow must be ejected from the system if a build-up of pressure is not to occur. If this bleed occurs downstream of the suit, it can be observed that some of the carbon dioxide which is generated will never get to the CO_2 remover. Therefore, the LiOH canister can be sized for a smaller CO_2 load in this type of system. Thus, with the exception of the oxygen tank, the weights of the components decrease due to the overboard flow. This decrease in weight, however, is negligible when compared to the gain in tank and oxygen weight required to maintain and store the primary flow. Cryogenic storage was not considered feasible to this application for a reason other than that mentioned above. The problem of recharging the bottle can be very significant. During the course of the LEM mission, the density of the cryogenically stored oxygen decreases. In order to obtain high density, cryogenic oxygen for the PLSS late in the LEM mission, some of the oxygen taken from the LEM tanks must be evaporated to cool the oxygen with which the PLSS tank is filled. Thus, an excess amount of oxygen over and above that required to fill the PLSS tank must be taken from the LEM storage tank. Whether or not this excess oxygen, which is boiled off, can be used is questionable. If part must be thrown away, this represents a weight penalty that should be charged to the PLSS along with any special tank filling apparatus. Moreover, if the PLSS oxygen bottle is sized for the high density cryogenic oxygen, any decrease in the density of oxygen that is finally achieved during a recharge will be reflected in less total stored oxygen weight, and consequently, less available mission time. The dangers in such a procedure are readily apparent. Chief among these would be the requirement of an extremely accurate oxygen quantity sensing technique in the bottle. With gaseous storage, pressure and temperature readouts are adequate to ensure the proper recharge.

Beside the ejector, this system is very similar to the fan motor system. A water boiler, water separator, and LiOH canister are required. It is possible that a supplemental heat exchanger could be used to some advantage in getting heat into the oxygen tank if cryogenic oxygen is used. This, however, is not achieved without some penalty in weight and reliability.

Figure 3 is a schematic of an oxygen motor powered system. This system is similar to the two previously described. The major difference is that the prime mover is a positive displacement oxygen powered motor. Oxygen enters the high pressure side of the motor and is exhausted overboard much in the same way as primary oxygen in the ejector system is bled overboard. On the low pressure side the system ventilation flow is pumped by a large piston. It is believed that an efficiency of 60 to 70 percent is achievable for the air pump.

The simplest system that could be envisioned is one based upon the use of a through flow oxygen flush system (Figure 4). In such a system, oxygen could be stored either in gaseous or cryogenic form. There is a pressure regulating valve to reduce pressure to the 3.5 psi operating level. The oxygen leaves the regulator and flushes through the suit and then overboard carrying with it all the oxygen, CO_2 and heat generated by the man. There is no need for any recirculation and, therefore, no need for any prime mover. Unfortunately, the amount of oxygen that would be required for a four-hour mission is prohibitive - amounting to 40 to 45 pounds. If a cryogenic storage tank is utilized, a reasonable estimate for tankage weight would be 40 to 45 pounds. Thus, without even considering miscellaneous support and valve components the system weight is over 80 pounds. This number gets successively worse when gaseous storage along with its accompanying higher tank weight is considered.

A novel, if not uncomplicated, method of PLSS design would be to incorporate a steam turbine which utilizes the energy in the steam exhausting from the water boiler (Figure 5). The operation of this system is basically the same as the fan motor system except that the fan is driven by a turbine rather than a battery powered motor. Theoretically this system appears at least feasible. A steam turbine design for this application was analyzed at the minimum metabolic load condition for which the steam flow exiting from the water boiler was minimum. The turbine pressure ratio was ten to one and the turbine rotor speed was 55,000 rpm. The required turbine efficiency to operate the fan and pump was found to be 40 percent, minimum, at a metabolic load of 800 BTU/hr. At minimum metabolic loads of 400 BTU/hr which may indeed be more reasonable, it was found that the system could not operate. Labyrinth seal and bearing losses were only estimated. In order to prevent the water boiler temperature from dropping below 32, a back pressure valve would be required between the boiler and turbine. Therefore, the inlet pressure available to the turbine at average operating conditions will be less than the water boiler exit pressure. It is questionable if the efficiency of this turbine could even approach 40 percent. It is believed that a reasonable efficiency for such a turbine is in the order of 20 to 25 percent. When compounded with the other disadvantages of complexity, reliability, etc., the turbine system was not considered to be a justifiable selection.

The system shown in Figure 6 is similar to the closed loop fan system with the exception that the oxygen supply is stored chemically as in potassium superoxide (KO_2). As the ventilation flow passes through the KO_2 canister, the water and carbon dioxide in the stream react with KO_2 to form oxygen. However, the reaction of KO_2 with CO_2 tends to be rather slow and an LiOH canister is still required for CO_2 removal.

The operation of this system depends on oxygen being produced as a function of the subject's sweat rate. Under most conditions, the latent load produced by the man will be adequate to produce the required oxygen. However, for low loads occuring simultaneously with negative heat leakage, it would be necessary to reduce the vent flow in order to maintain sweat production. While this could be accomplished automatically, it is possible that the resulting flow may be inadequate in terms of CO_2 purging effectiveness.

Within each system the weight and volume of each major component was estimated. A system reliability number was then determined based mostly on existing failure records of components similar to those used.

The fan motor system was chosen for development because the predicted system weight and volume were the least for all systems investigated and because its success did not depend on any significant improvement of existing technology. Moreover, the fan system appeared to have the best growth potential. Improvements in fan, motor, or battery performance will yield a smaller back pack. Without increasing the back pack size, increases in metabolic rate capacity or mission duration capacity were projected. The system powered by an ejector had the highest predicted reliability number of all the systems only when gaseous oxygen storage was considered. However, the difference between reliability between the fan and ejector systems only appears in the forth place after the decimal.

The air motor system is limited due to the volume required for oxygen storage in the same way as the ejector. While the oxygen motor and pump assembly did not require any advance in technology, there was not sufficient experience with them upon which to base a system. The predicted weight and volume of the chemical and oxygen storage system was similar to the fan system. However, the dependence upon sweat production to insure oxygen availability was unattractive. If a supplementary gaseous oxygen supply were used to circumvent this feature, there no longer was a valid reason for any chemical oxygen storage.

OPTIMIZATION OF SELECTED SYSTEM

When the fan motor battery system was selected, a computer program was written for the IBM 7090 computer in order to facilitate the solving of hundreds of design point cases. The effect of system flow rate and metabolic heat load on system weight, power, and volume was plotted. Figure 7 is a typical set of parametric curves determined from computer output. In it can be seen quite graphically the strong influence that metabolic load has on required ventilation flow and total system heat load. This curve is plotted at a value of system heat leak (inward) of 100 BTU/hr. A fan efficiency of 65 percent and a motor efficiency of 60 percent was used in this trail. An effectiveness of mass transfer of 2/3 applies to the entire family of curves. This effectiveness of mass transfer is defined as that portion of the suit exit volume flow which leaves the system saturated while the remainder leaves at inlet dew point. While it is known that this is a fictitious situation and that moisture picked up within the suit by simultaneous heat and mass transfer is more or less evenly diffused in the stream, the concept of effectiveness of mass transfer proved to be a useful tool with which to analyze this portion of the loop. Figure 8 shows the effect of effectiveness on the volume flow rate required in the system as a function of metabolic heat load and heat leak. Note that for very poor values of effectiveness a much higher flow rate is required to remove this same amount of heat. Therein lies the fundamental basis for desiring a high value of mixing between the ventilation flow and the vapor sweated by the man. If this ventilation flow could leave the suit loop saturated, then a minimum value of flow could be achieved. Figure 9 shows the pressure across the fan that would be required to pump the flow commensurate with the metabolic loads and mass transfer effectiveness indicated. The significance of this curve can only be assessed when the dependence of power on pressure rise is shown. This motor power as a function of required fan ΔP is shown in Figure 10.

SYSTEM EVALUATION

Upon completion of the fan battery prototype system, a program for testing it both unmanned and manned was planned. Before commiting a man to the system it was, of course, necessary to evaluate some of its more important performance characteristics. A facility (Figure 11) nicknamed, "The Canned Man" was established which was designed to simulate typical metabolic outputs of man. It, therefore, consumed oxygen, generated carbon dioxide, generated a latent sweat load, and a sensible load. It simulated suit pressure drop at any value that was predetermined. In this way, the effect of various pressures and size occupants in a suit as they effect pressure drop could be simulated. All the parameters mentioned here were variable by the operator. The oxygen supply in the charged PLSS bottle was "consumed" by varying the opening in a manually controlled valve; the flow through this valve having been previously determined as a function of the metabolic load that was being simulated. Similarly, a predetermined value of carbon dioxide could be fed into the loop eventually finding its way to the LiOH canister. In this way canister performance as a function of CO_2 loading could be evaluated. The canned man and prototype development back pack were heavily instrumented so that a complete set of data on all component performance could be obtained in addition to the system performance characteristics.

Figure 12 is a plot of the corrected system pressure drop as a function of system ventilation flow rate. The pressure flow characteristics were taken at the minimum and maximum position of the flow control valve setting. Figure 13 shows the PLSS pressure rise characteristics which is the net rise of the fan minus the losses within the back pack. Therefore, the solid curve in this figure represents the pressure rise across the back pack if the back pack were considered as a single component (or fan). The dotted line in this figure represents the predicted space suit load line when pressurized and manned. The intersection of the two curves then yields a predicted operating point. This point could only be proved valid when the back pack and a pressurized suit were mated. When this test was actually accomplished, it was found that the predicted operating point was achieved within 5 percent.

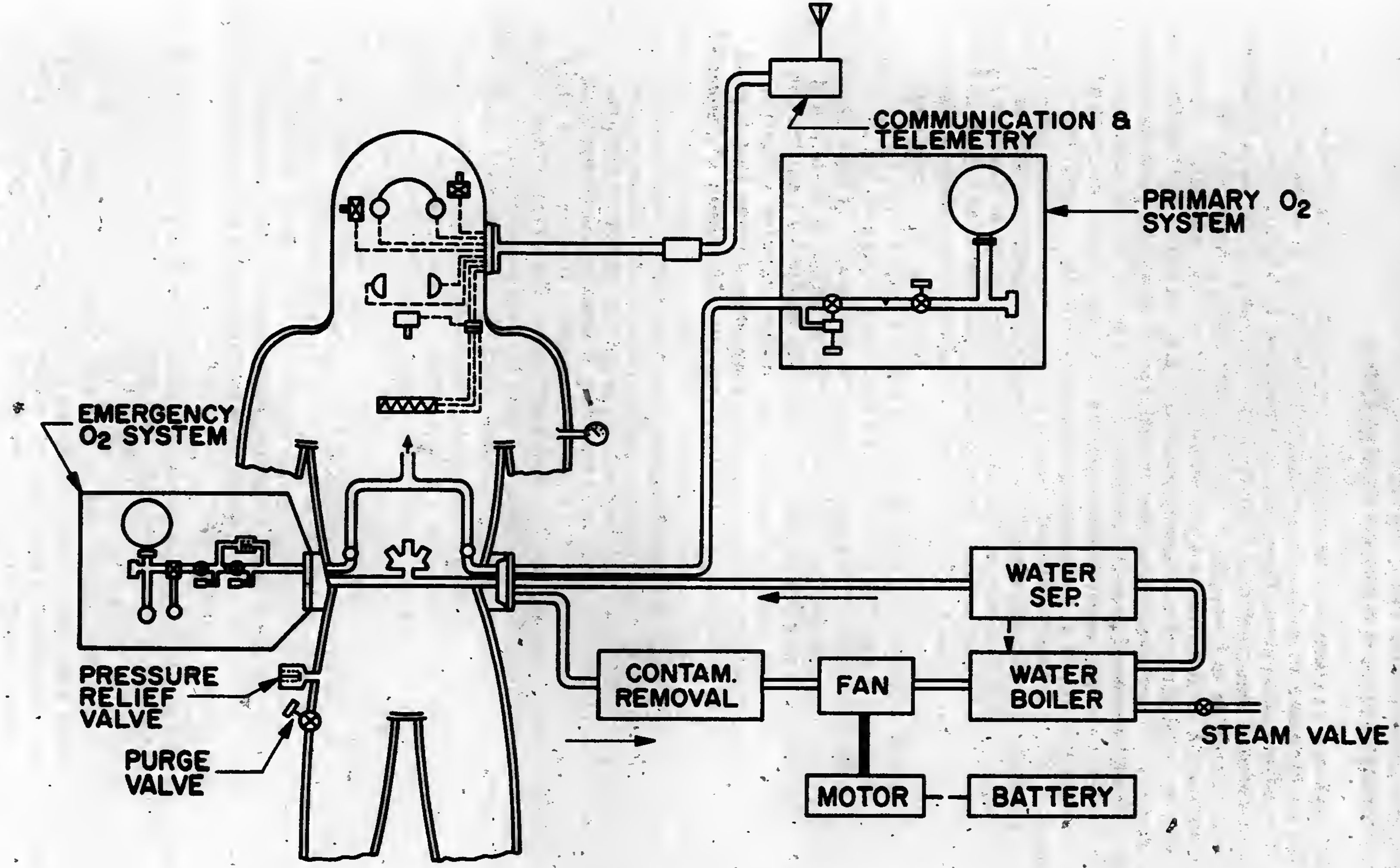

Figure 1

PLSS Schematic Concepts
Prime Mover - Fan Driven by Battery-Powered Motor
Ventilation Loop - Closed

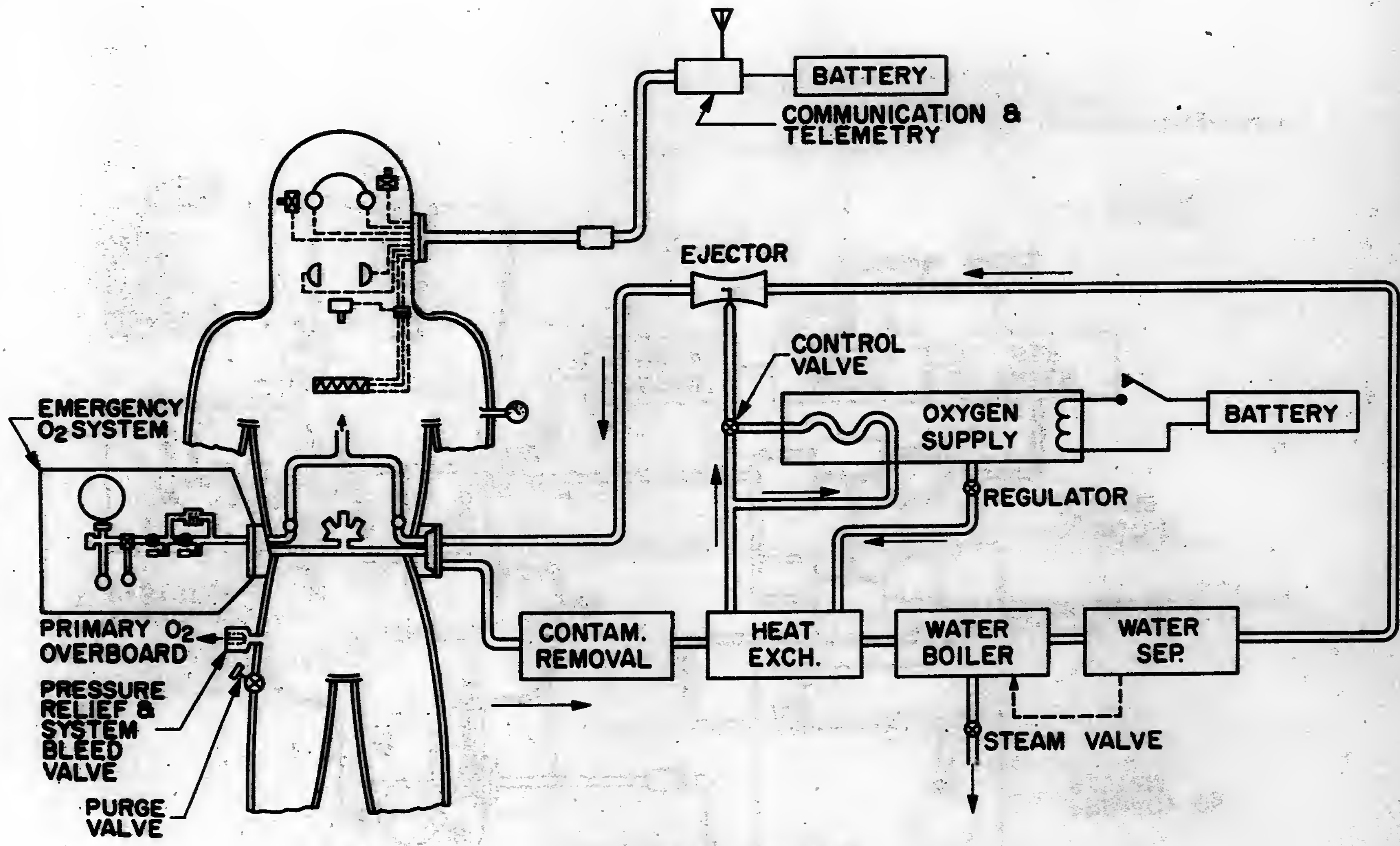

Figure 2

PLSS Schematic Concepts
Prime Mover - Ejector
Ventilation Loop - Semi-Open

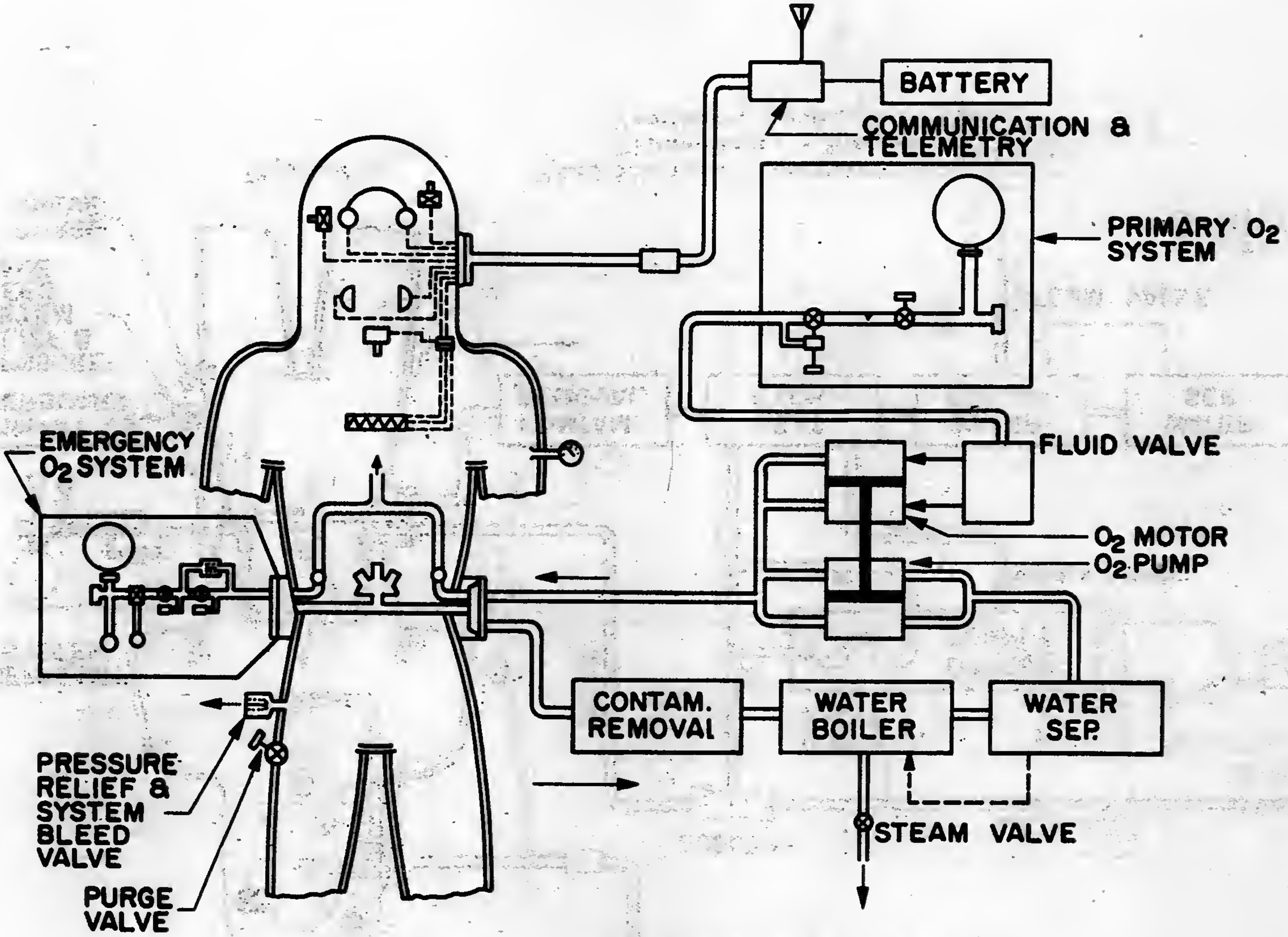

Figure 3

PLSS Schematic Concepts
Prime Mover - Positive Displacement Oxygen Motor and Pump Assembly
Ventilation Loop - Semi-Open

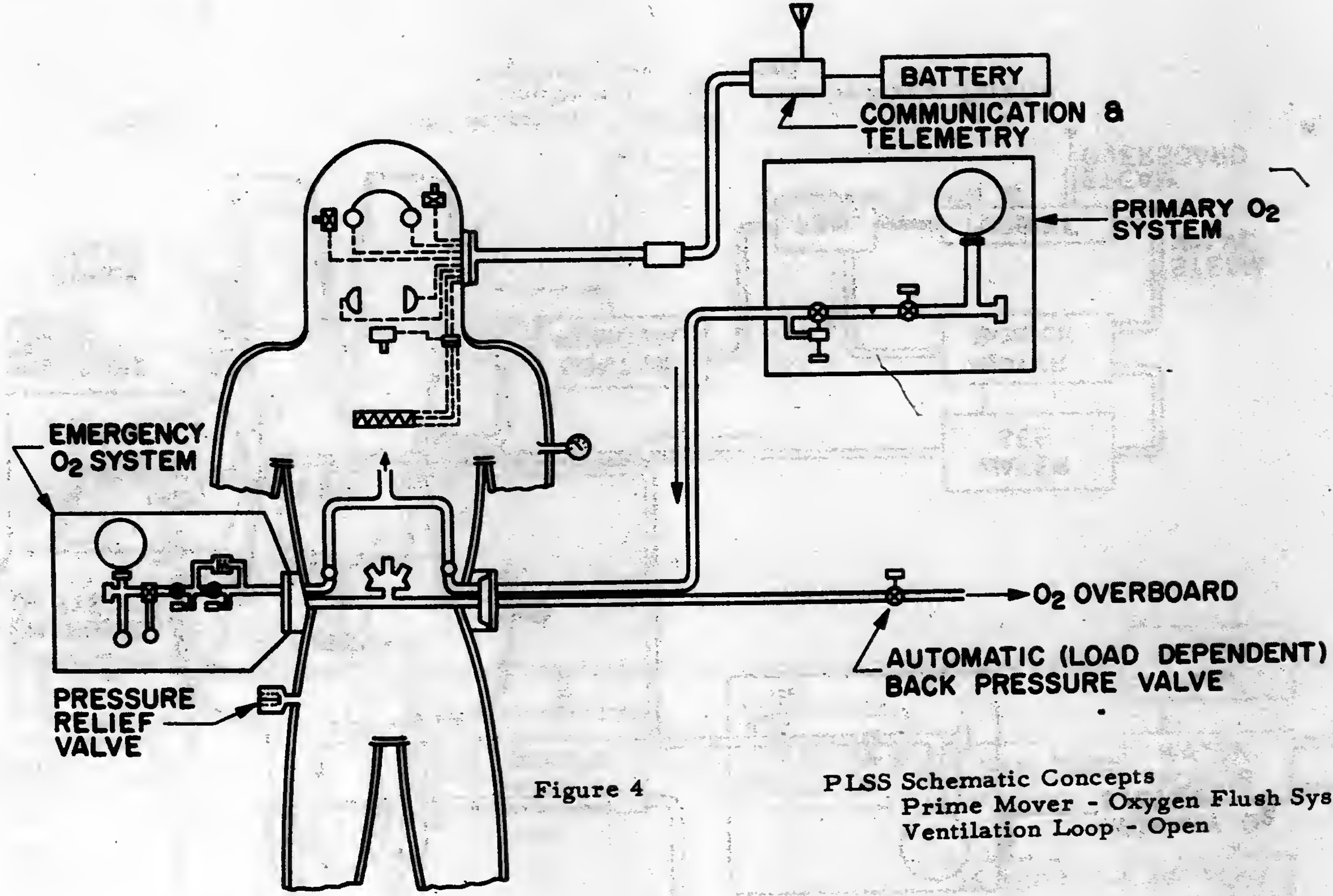

Figure 4

PLSS Schematic Concepts
Prime Mover - Oxygen Flush System
Ventilation Loop - Open

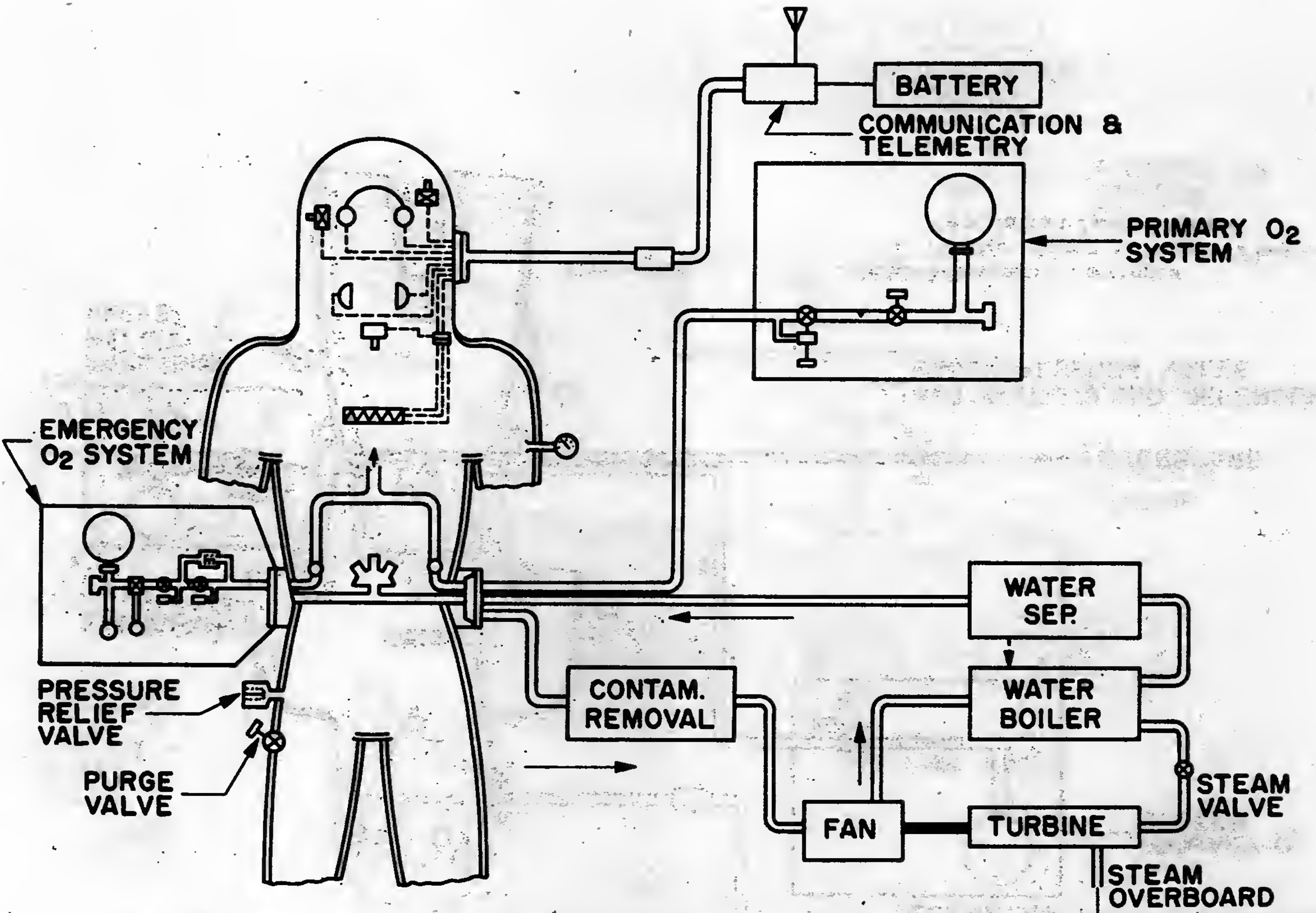

Figure 5 PLSS Schematic Concepts
Prime Mover - Fan Driven by Steam Turbine
Ventilation Loop - Closed

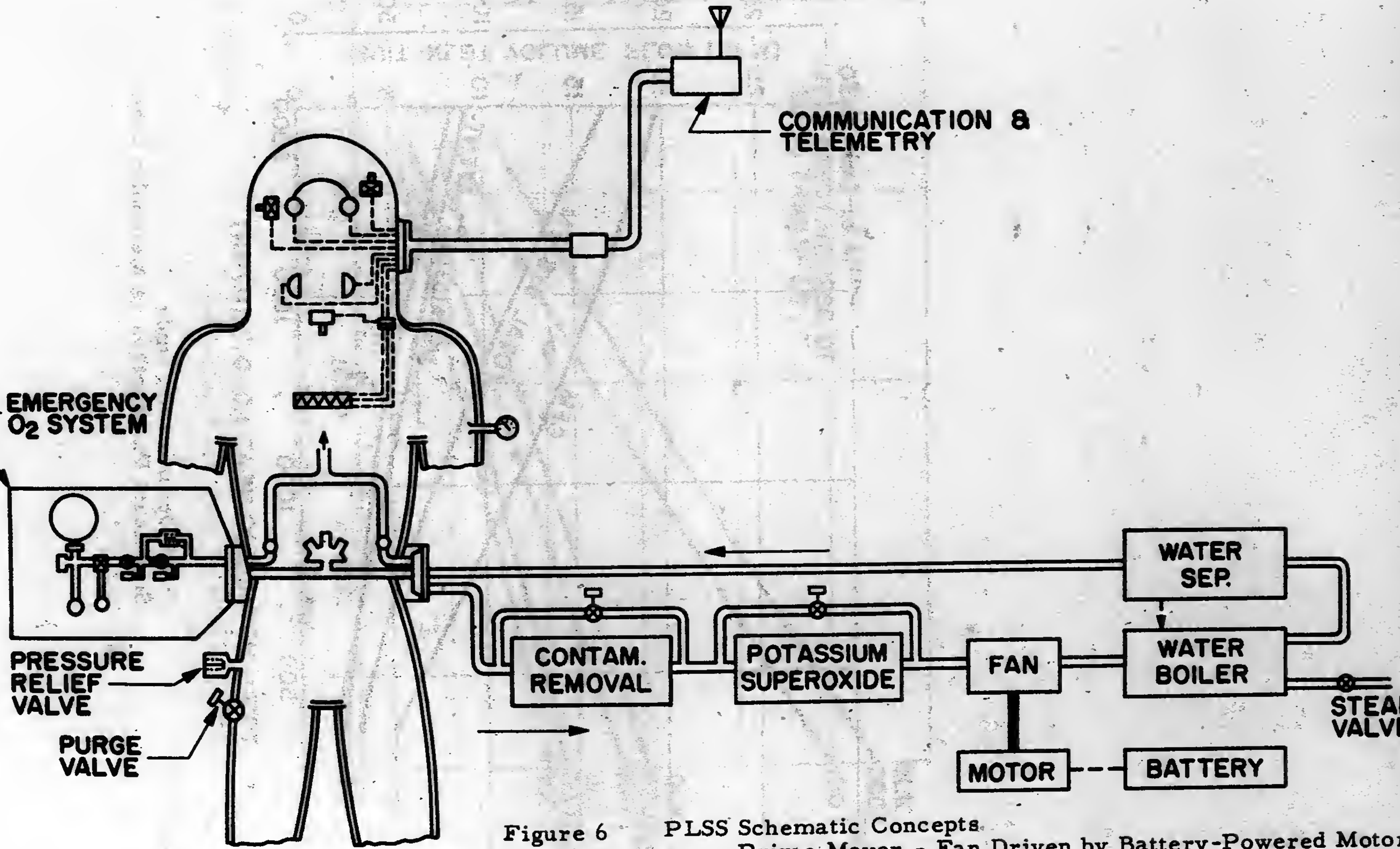

Figure 6 PLSS Schematic Concepts
Prime Mover - Fan Driven by Battery-Powered Motor
Ventilation Loop - Closed; Chemical Oxygen Storage

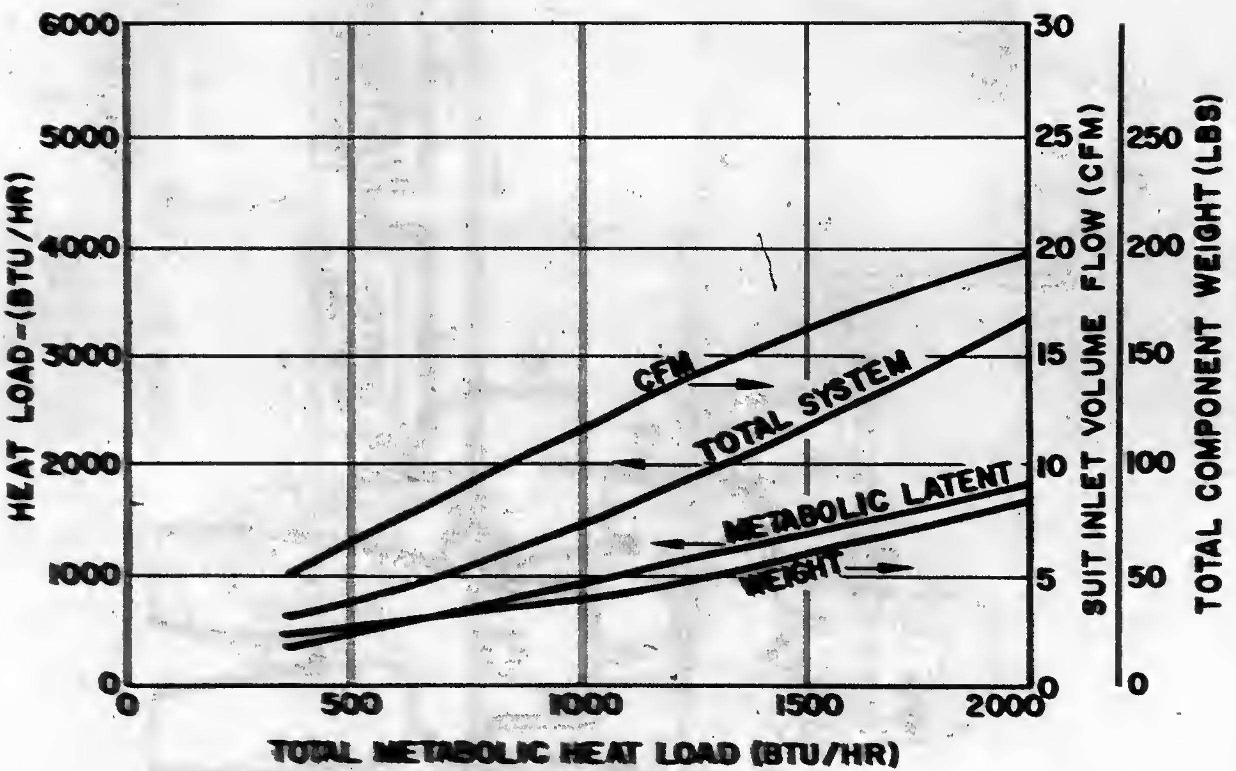

Figure 7 PLSS Optimization Study
Significant Parameters vs Metabolic Load

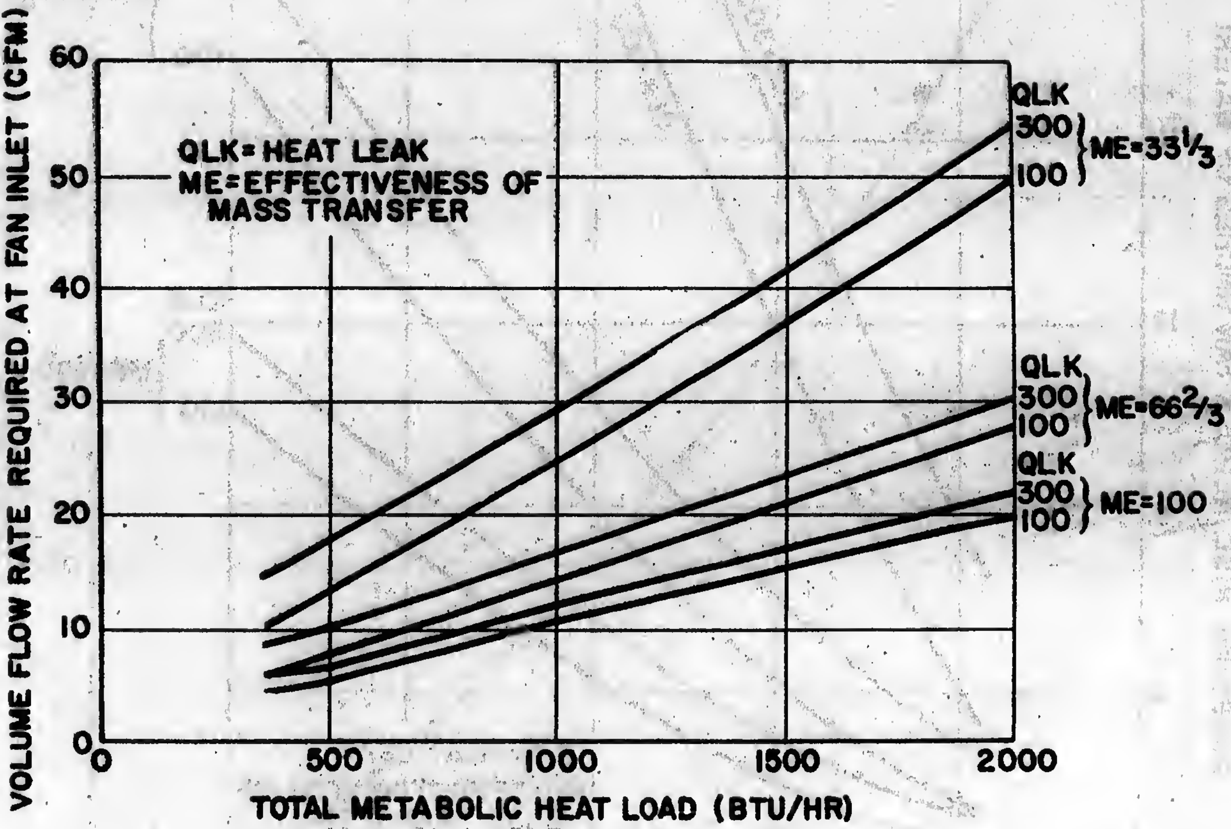

Figure 8 PLSS Optimization Study
Ventilation Flow Rate vs Metabolic Load

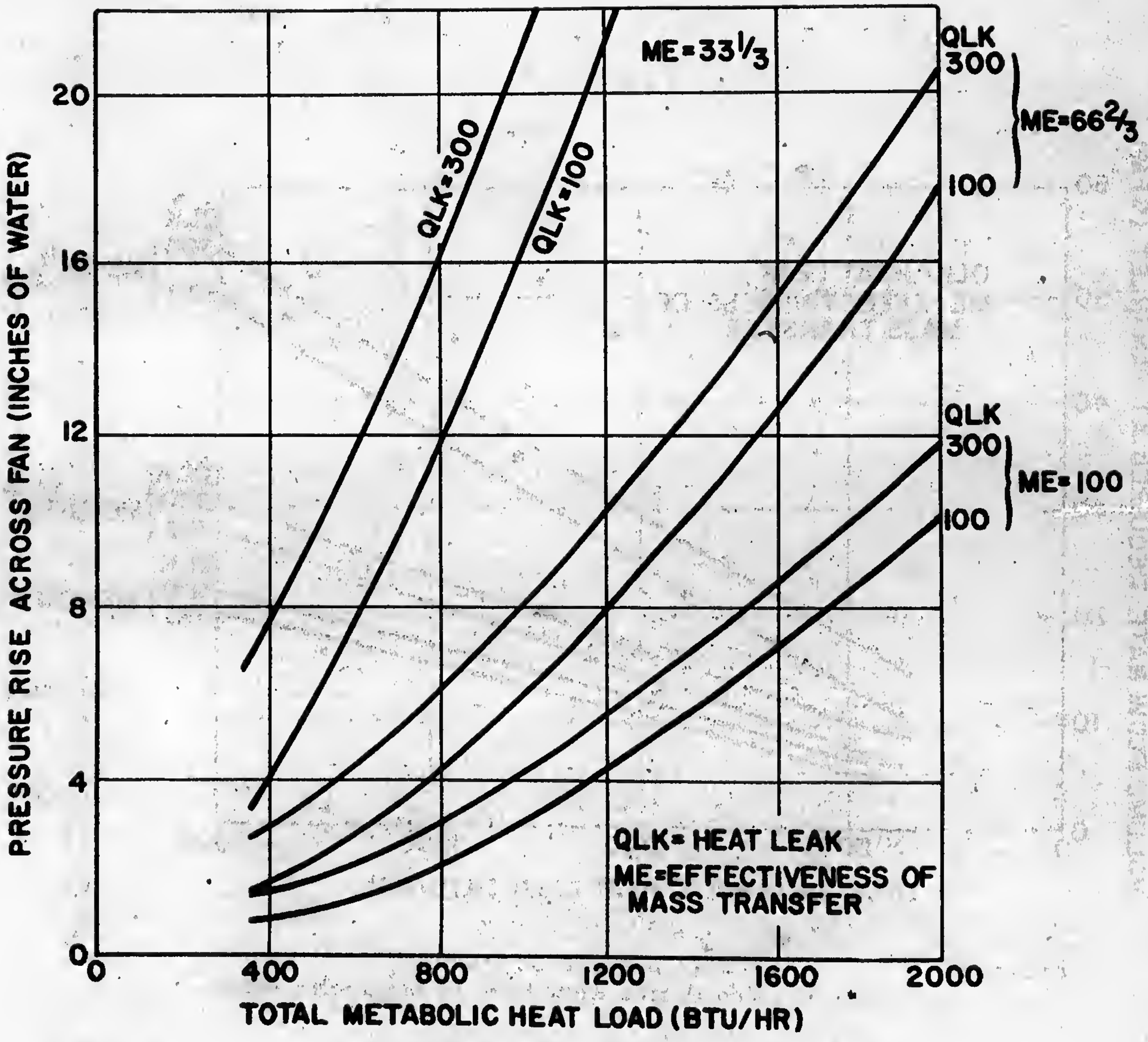

Figure 9 PLSS Optimization Study
Required Fan Pressure Rise vs Metabolic Load

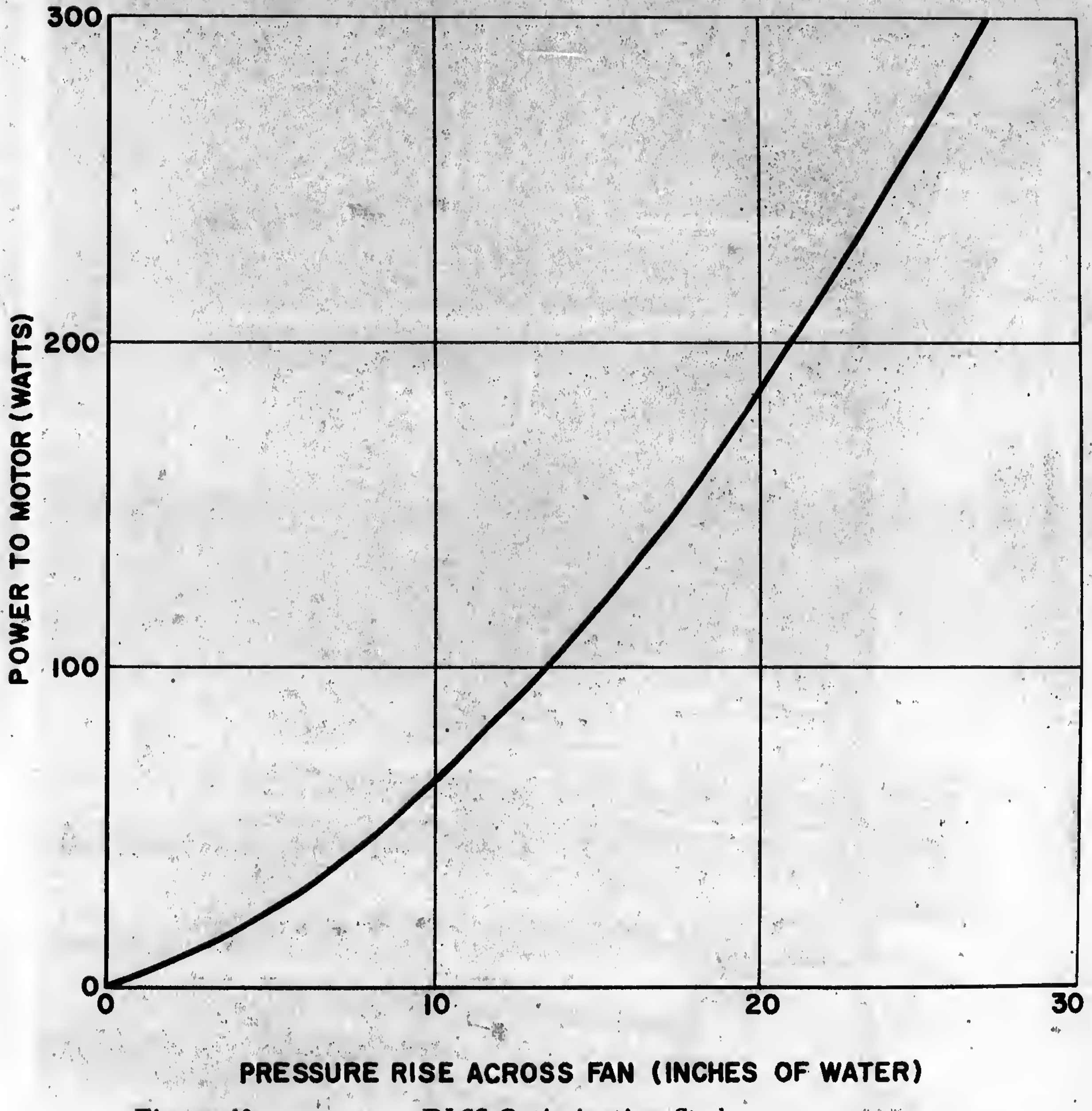

Figure 10 PLSS Optimization Study
Motor Power as a Function of Fan Pressure Rise

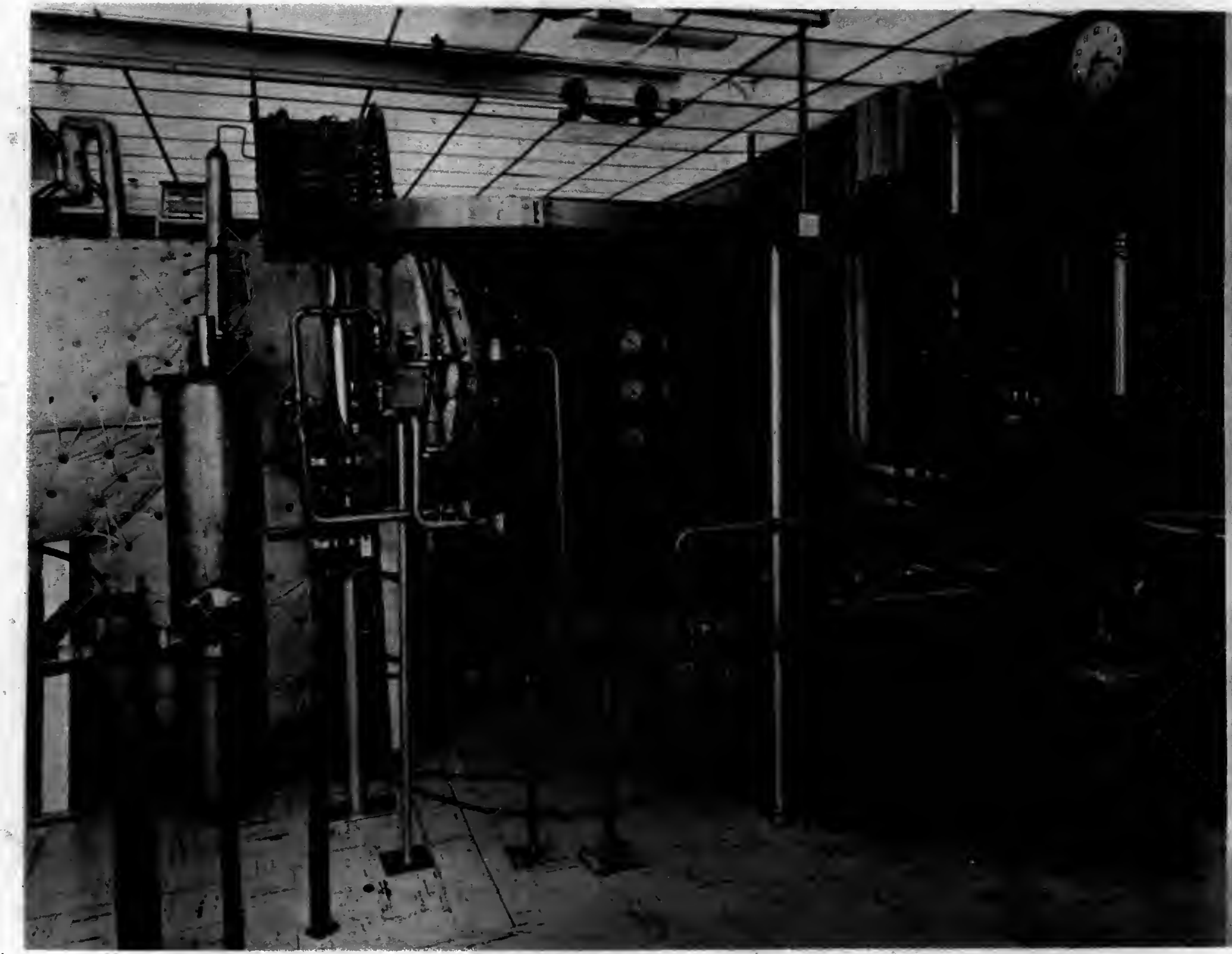

Figure 11 "Canned Man" Test Facility for PLSS Performance Testing

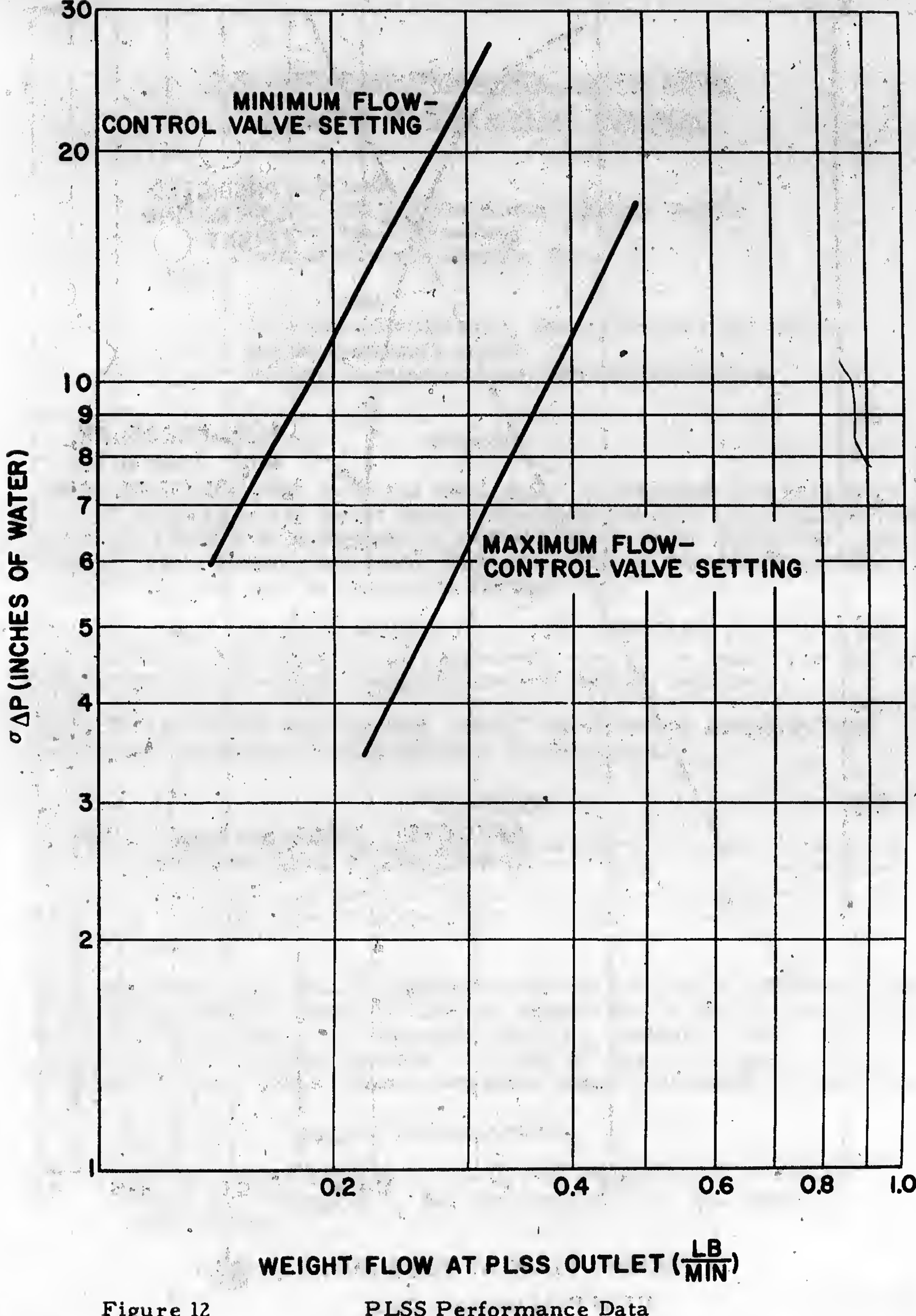

Figure 12 PLSS Performance Data
Ventilation Flow vs Corrected Pressure Drop

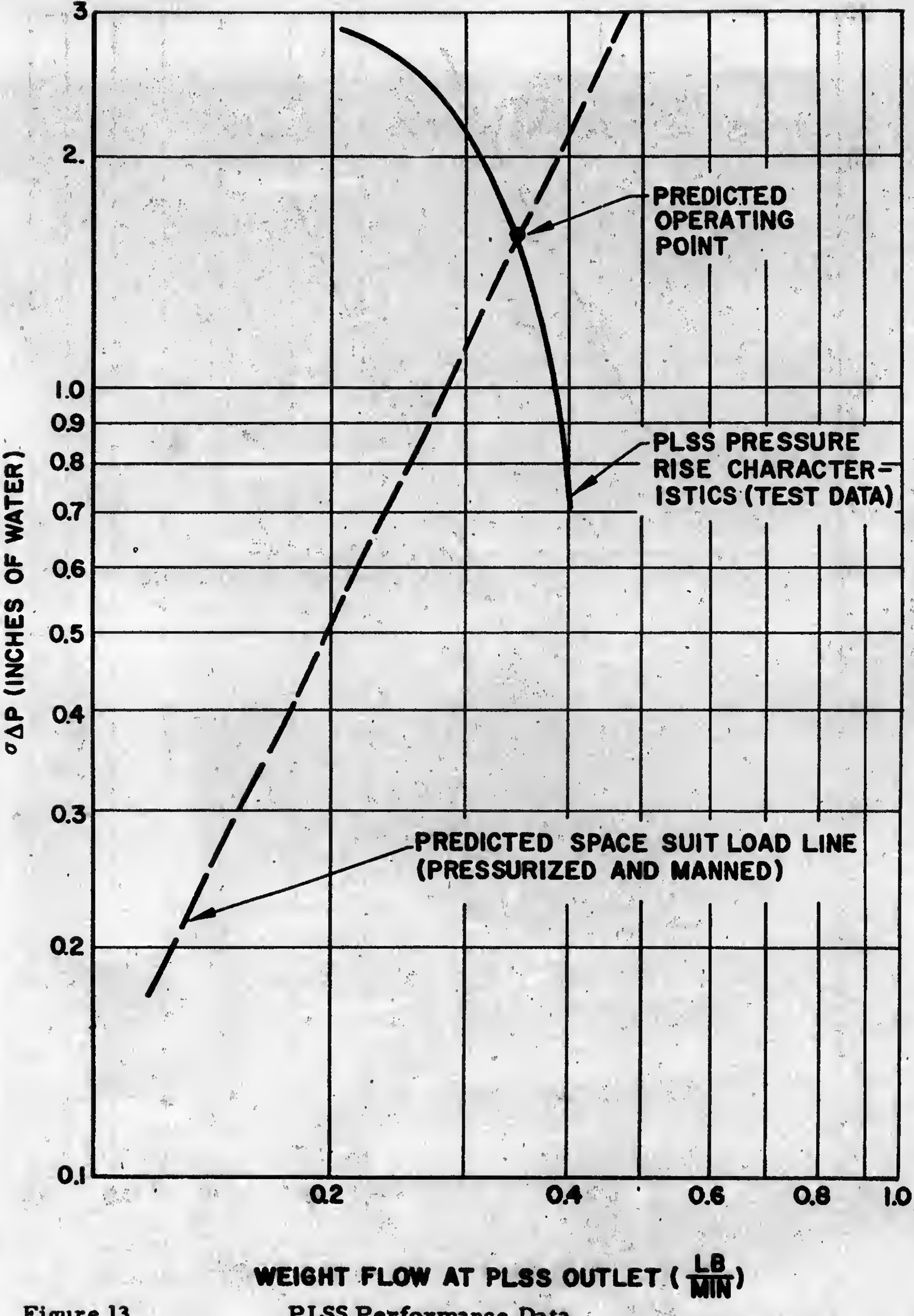

Figure 13 PLSS Performance Data
Net Backpack Pressure Rise and Suit Pressure Drop vs Ventilation Flow

LIFE SUPPORT AND ENVIRONMENTAL CONTROL SYSTEM FOR A LARGE ORBITING RESEARCH LABORATORY

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SUMMARY

Recent studies for both large and small space laboratories have provided a fund of information and design data. This paper presents an evaluation and the selection of a life support and environmental control system for a large "zero g" space research laboratory that is capable of supporting man and equipment for one to five years with resupply.

A conditioned shirtsleeves environment of 70°F temperature and 7 psia pressure is normally provided to all compartments and to the hangar area during the loading and unloading of exploration and logistic vehicles.

The life support and environmental control system weight including fixed weight and consumables is approximately 36,000 pounds.

Introduction

Recent studies for both large and small space laboratories have provided a fund of information and design data. A typical "Post 1968" research laboratory investigated and shown in figure 1 was 33 feet in diameter and 100 feet long; launched by a two stage Saturn C-5 (S-1C and S-II) into a 260 n.mi. circular earth orbit.

This manned orbiting research laboratory must be provided a life support and environmental control system (LS/ECS) to support the 24 man crew complement and to condition laboratory equipment. Figure 2 presents a functional block diagram of the subsystems required to accomplish the life support and environmental control task. These subsystems provide the following functions:

- Atmospheric Supply and Pressurization

  Storage and delivery of N_2 and O_2 for replenishment of leakage losses and storage of O_2 and regeneration of O_2 for metabolic requirements.

- **Atmospheric Conditioning and Purification**

  Humidity control and CO_2, trace contaminant and particulate removal and pressure suit conditioning.

- **Thermo-Conditioning**

  Temperature control of the compartments, pressure suits, and equipment; heat source for water, food heating, CO_2 silica-gel bed desorption and urine recovery.

- **Water and Waste Management**

  Storage and supply and reclamation of water from urine, perspired and expired water, treatment of fecal wastes.

Figure 3 shows the general location of the major life support and environmental control equipment. Since most of the crew members are usually in the crew quarters and laboratory, location of a life support system in each of these compartments permits convenient accessibility and minimizes distribution power requirements. In addition, the half-systems are located in two separate pressure vessels in case of a fire or other emergency. This is considered to be an extremely remote possibility. Figure 4 presents a laboratory cross section that shows the relative location of a half-system and the distribution ducts routed through the central traffic tunnel. Twelve pressure suit stations are available that each has umbilicals to support six men. Each pressure suit station can operate either independently or with the cabin atmospheric conditioning and purification subsystem.

The design conditions together with the trade-offs and a proposed integrated subsystem preliminary design are presented.

Design Conditions and Assumptions

Complete redundancy of all life support and environmental control equipment is provided. Each of two half systems normally supports 12 men, but each has the design capability of supporting the entire 24-man crew. In addition, 12 space suit stations are provided that each will accommodate six men.

Other design conditions and assumptions are summarized below.

Mission

Mission Duration	1-5 years
Orbital Altitude	260 n. mi.
Initial LS/ECS Supply	180 days
Resupply Period	90 days

Vehicle

Pressure Compartment Volumes

Laboratory	20,090 ft^3
Crew Quarters	17,130 ft^3
Storage	13,970 ft^3
Hangar	24,900 ft^3
Central Tunnel	2,230 ft^3
Airlocks	600 ft^3
Total	78,920 ft^3

Leakage Rate	17 lb/day

Atmospheric Parameters

Temperature	$70 \pm 5^\circ$F
Total Pressure	7 psia
Oxygen Partial Pressure	3.5 psia
Diluent	Nitrogen
Carbon Dioxide Partial Pressure	0.15 psia maximum
Relative Humidity	35 to 50%

Crew

Crew Size

Normal	24 men
Emergency	36 men

Metabolic Parameters

Oxygen Consumption	1.84 lb/man/day
Carbon Dioxide Output	2.12 lb/man/day
Water Consumption	6.16 lb/man/day
Wash Water	40.00 lb/man/day
Urine	3.41 lb/man/day
Metabolic Water	0.67 lb/man/day
Respiration & Perspiration	3.20 lb/man day
Water in Fecal Output	0.33 lb/man/day
Food Intake	1.30 lb/man/day
Heat Output, Latent	3,340 btu/man/day
Heat Output, Sensible	7,860 btu/man/day

Electrical

Electrical Heat Load	27 kw
Voltage	28 ± 10% V.D.C.
Power Penalty	450 lb/kw.

Atmospheric Supply and Pressurization Subsystem

The atmospheric supply and pressurization subsystem provides the storage and delivery of N_2 and O_2 for replenishment of leakage losses together with storage of O_2 and regeneration of O_2 from expired CO_2 for metabolic requirements. Also included are the leak detection and airlock/hangar decompression units.

The discussion of atmospheric supply and pressurization subsystem selection and operation together with the parametric trade-off analysis and evaluation, is given below.

Selected Subsystem

A combined system consisting of subcritical storage of oxygen and nitrogen and Bosch oxygen regeneration is recommended for the laboratory atmospheric supply. Trade-offs shown in figure 5 indicate that oxygen regeneration is advantageous for missions longer than 5 months. Furthermore, the resupply penalty is reduced by approximately 16,000 lb/yr.

Subcritical storage of oxygen and nitrogen was selected because it is clearly the lightest in weight, as indicated in figure 6. Test results in the last two years for space application have indicated no major problems with subcritical storage. A comparison of major advantages are listed below:

- Subcritical Storage

  Lighter weight

  Simpler transfer, filling and top-off

- Supercritical Storage

  Operational and proven

  Simpler to measure quantity in tank

  Less plumbing required.

The Bosch-type regeneration unit is proposed for O_2 recovery because it is considered the furthest along in development and provides the least overall weight penalty. A flow diagram of this unit is given in figure 7. The major components are the reactor, carbon collector, heat exchanger, condenser-separator, and water electrolysis cell.

Atmospheric Storage Subsystems Integration

Figure 8 is a schematic for the storage system. It includes subcritical storage of nitrogen and oxygen and provision for compressing and storing relatively small quantities of high-pressure gas used for charging backpacks and nitrogen used for refilling fire extinguishers. Pressurized nitrogen is also supplied to water storage tanks, reservoirs, and accumulators to provide positive expulsion for zero-g delivery. Three oxygen tanks and two nitrogen tanks, of approximately 3 ft-diameter each, are adequate to replenish predicted leakage losses and to provide make-up metabolic oxygen for 180 days.

The oxygen and nitrogen boiloff must be vented when it exceeds the design use rate, and hence represents wasted gas. A practical design limit, where insulation does not become prohibitive, is a design use rate of one pound per day per tank, which is well below the predicted use rate necessary for leakage replenishment and metabolic oxygen supply. Design for a lower boiloff rate than use rate allows a savings in venting losses if the 17 pound per day leakage estimate is higher than actual leakage, or if crew size is less than design. The electrical power or waste heat required amounts to only about one watt for each pound per day delivered above the design use rate.

Figure 9 shows a schematic of the atmospheric storage and distribution system within the vehicle. The tanks, located in the hangar, minimize boiloff during resupply. Oxygen and nitrogen are fed as needed into the atmospheric supply ducts to the several pressurized compartments. Oxygen can be supplied upon demand to each of the twelve space suit stations. Nitrogen can be supplied to positive expulsion systems within storage tanks, and to fire extinguishers. An extra tank of subcritical oxygen, not shown in figures 8 and 9, is located in the storage compartment. This supply can be used for normal operation but also provides the last five space suit stations, 8 to 12, independently of the primary source located in the hangar. This supply would be available if an explosion should occur in the hangar area and the crew was required to wait for a rescue vehicle to dock to the external port.

Hangar and Airlock Pumpdown

During loading and unloading operations, a 7 psia shirtsleeve environment will be maintained in the hangar. At other times, 1.0 psia will be provided to minimize leakage and to allow storage of the logistics and exploration vehicles for extended time periods. As indicated in figure 10, pumpdown is desirable for more than four logistics operations.

The hangar, when pressurized to 7 psia, contains approximately 1,040 lb of atmosphere. Expenditure of this quantity of air, even once every 90 days, presents a significant weight penalty. Therefore, provisions were made for pumping and high pressure storage of the hangar atmosphere. Pumping the hangar atmosphere back into the crew pressurized compartments would raise the cabin pressure and might introduce harmful contaminants. From the viewpoints of crew safety (toxicity) and experimental control, neither of these consequences was considered tolerable. The pumpdown unit is driven by a 4 HP motor. The unit may also be utilized for airlock evacuation. Approximately 11.8 days would be required to pumpdown the hangar to 0.1 psia. At this final pressure, only 15 lb of atmosphere need be exhausted to space. The airlock and hangar pumpdown system is shown in figure 11.

Leakage and Leak Detection

A summary of the leakage criteria and assumptions for LORL are tabulated below:

Leakage Source	Order of Magnitude
Hatches and hangar door	7×10^{-3} lb/day/in. (7 psia)
Meteoroid penetrations	Complete loss of the atmosphere in one pressurized compartment when punctured
Electrical leads	10^{-7} lb/day/in. (negligible)
O-ring seals (replaceable)	10^{-9} lb/day/in. (negligible)
Valves	10^{-3} lb/day/valve (negligible)
Seam and joints (welded)	10^{-14} lb/day (negligible)
Diffusion through vehicle skin	10^{-14} lb/day (negligible)

The only leakage that is considered appreciable is that through hatches or hangar seals and through holes caused by meteoroid penetrations. The assumed seal leakage (7×10^{-3} lb/day/in) is based on MIL STD S-8484 for gaskets. The meteoroid penetration leakage assumption of complete loss of the atmosphere in a pressure vessel per puncture is considered to be quite conservative.

Based on the above criteria, total vehicle leakage is estimated to be approximately 17 lb/day, of which 4.75 lb/day is due to seal leakage and 12.25 lb/day allowed for back-up in the event of inadvertent leakage or a serious meteorite puncture. Considerable oxygen and nitrogen should be available to supplement the 90-day reserve if the leakage is less than 17 lb/day.

Leak Detection

Separate detection is provided for leaks caused by meteoroids and by failure of cabin seals. Each cabin seal is monitored separately by an external sensor to assure seal effectiveness. If the seal fails, replacement without cabin decompression is made possible by a dual seal arrangement. Compartment leakage is detected by the monitoring of the nitrogen flow from the supply system. A change in supply rate over a significant period of time indicates the presence of small leaks.

Crew Safe Time

Due to a multi-pressure vessel concept, the need to rapidly repressurize any one compartment is unlikely. In addition, decompression of large volumes is, in most cases, slow. Consequently, considerable time for corrective action is available. Figure 12 indicates the decompression time to the crew danger point (i.e., when oxygen partial pressure falls to 2.4 psia) as a function of pressure vessel volume. For the crew quarters compartment (approximately 17,000 cu.ft.), almost three hours are available even in the unlikely event of a half-inch-diameter penetration. During this time, the crew may don pressure suits or transfer to alternate pressurized compartments.

When a leak is repaired, repressurization directly from the subcritical tanks may be accomplished slowly at a minimum power drain. For the 17,000-cu.ft. crew quarters compartment, 1 kw of power is required to repressurize in 16 hours for the extreme case of a near-zero pressure to 7 psia.

The large volumes of the LORL pressurized compartments provide a safeguard against catastrophe due to oxygen supply system failure. Figure 13 indicates the safe time after supply shutdown as a function of pressure vessel volume and the number of crew members. Even if the entire 24-man crew is located in the crew quarters compartment, more than 55 hours are available for corrective action. The capability of each of the two oxygen regeneration units to support the entire 24-man crew further increases safety and reliability.

Atmospheric Conditioning and Purification Subsystem

The atmospheric conditioning and purification subsystem provides humidity control and carbon dioxide removal together with monitoring and removal of harmful trace contaminants that may be generated by the crew and equipment. Discussion of the pressure suit loop integration is included in this section.

Atmospheric conditioning and purification subsystem selection and operation and the parametric trade-off analysis and evaluation is discussed below.

Selected Subsystem

Figure 14 shows the component arrangement and flow paths for one of the two identical atmospheric purification units.

The major components included in each unit are a condenser and wick-type water separator and silica-gel for humidity control, a pair of molecular sieves for carbon dioxide removal, and a catalytic burner for trace contaminant control. Also included are a debris trap, charcoal-bed odor remover, chemisorbent bed, fine filter, and ultraviolet lights.

Investigation of carbon dioxide removal units indicates that a molecular sieve type unit is the furthest along in development and is proposed. However, there are several drawbacks to this system. Molecular sieves, or zeolite beds, have a preferential affinity for water over CO_2, which necessitates that almost absolutely dry air enter the beds. Silica-gel beds used upstream of the molecular sieves have a large influence on the LS/EC system. It requires a compromise in the subsystem design and complicates the subsystem valving and control systems. A nonmoisture-sensitive CO_2 absorber would simplify the CO_2 removal system, increasing reliability and allowing more flexibility in system design. A solid amines absorber being tested shows some promise. Desorption of a solid amines absorber requires heating at 170°F and evacuation to 25 mm Hg abs. The total penalty comparison between CO_2 removal units is given in figure 15.

Figure 16 shows that either atmospheric purification unit is capable of supplying all pressurized compartments and airlocks. In addition, the aft unit may directly support the hangar while the forward circuit supports the rest of the vehicle during loading and unloading. Due to the possible introduction of harmful contaminants from the logistics vehicles, it is not considered advisable to mix these two "air" streams. Additional catalytic burners are provided in the hangar to reduce this contaminant hazard.

Laboratory atmosphere monitoring is provided by a gas chromatograph and quadrupole mass spectrometer. The gas chromatograph is used for periodic sampling of the major atmospheric constituents and critical contaminants; the spectrometer is used for greater sensitivity in identifying any unknown contaminants that appear in the atmosphere.

If the carbon-dioxide removal system malfunctions, figure 17 indicates that approximately 11 hr are available for corrective action, even if the entire crew is located in the crew quarters.

Pressure-Suit Integration

Under certain modes of normal and emergency operation, the crew must use pressure suits. Life support backpacks are also required for additional protection when tasks are performed in remote laboratory locations.

Figure 18 illustrates the recommended method of pressure-suit integration into the vehicle LS/ECS. The suit loop shown has its own fans for circulation, condenser and water separator for temperature and humidity control, an activated charcoal filter for odor removal, control systems and suit connections. Each loop can operate either independently or with the cabin atmospheric control and purification subsystem. The suit loops operate independently by recirculation of 100% oxygen and leakage of a controlled amount for carbon dioxide partial pressure control. This leakage rate is approximately 1.5 lb/man/hr. Under normal operation, the bleed mode allows independent operation of each suit loop without extra carbon-dioxide removal units. This system provides backup support to the cabin LS/ECS during emergencies; the backpack system may be saved for an extreme emergency situation, such as rescue. The system has been designed to accommodate the Apollo spacesuit and backpack.

Thermo-Conditioning Subsystem

Cooling is provided to remove the metabolic heat of the crew, heat generated by the life support equipment and electronic equipment, and heat generated by the radioisotope heaters. Cooling is also provided for water, food and experimental apparatus. Heating is required for water, food, operation of the carbon-dioxide removal units and water recovery units, and for experiments.

Figure 19 shows that 22 sheets (1/2 inch thickness) of high-performance insulation is required to maintain the internal wall temperature above 65°F dewpoint for the anticipated values of solar absorptivity-long wave emissivity ratio (α/ε), and internal heat transfer coefficient (h_c). The thickness of foam-type insulation necessary for internal wall temperature control would be prohibitive for the zero-g LORL. Figure 20 presents the heat loss from the vehicle to space for 1/2 inch of HPI.

Space Radiators

The space radiators consist of parallel rows of tubing mounted on 4-inch centers inside the load-carrying structure. Every other row of tubes is utilized for active heat rejection; rows in between are available for backup. For the expected average fluid temperature in the radiators (80°F), approximately 60 sq-ft of radiator/kw of heat rejection is required as shown in figure 21 for the selected configuration 2. The internal heat load from the crew and equipment, 33.3 kw, less the computed loss through the vehicle wall of about 0.7 kw, results in a net heat rejection load of 32.6 kw. The resulting radiator area is approximately 1,950 sq-ft.

Location of the radiator tubes as shown in configurations 2 and 3, requires a larger number of tubes and area, but permits better accessibility and nearly four times the meteorite protection as compared to mounting to the inner side of the external skin (configuration 1).

Redundancy in radiator circuitry and pumps is considered necessary because the safe time to the crew danger point after pump failure or loss of coolant is quite limited. If all 24 men are located in the crew pressurized compartment and the equipment heat load is 8 kw, less than 22 minutes are available before an effective temperature of 90°F is exceeded.

Cooling and Heating Loops

The recommended method of cooling and heating is by providing two separate loops, as indicated in figure 22. In the cooling loop, FC-75 fluid is pumped through the various components. The heat absorbed is then rejected to space by the radiator. The major advantages of FC-75 are its low freezing point (-80°F) and its high dielectric constant; the latter makes it suitable for direct use in electronic equipment. The heat required in the life support subsystem is supplied by two Promethium 147 isotope heaters, each with a thermal capacity of 1.5 kw. This isotopic heat source saves 1,050 pounds and 5.5 million dollars over electrical heating from a solar cell-battery system.

Water and Waste Management Subsystem

The water and waste management subsystem provides for collection of urine and feces; reclamation of wash water, atmospheric condensate, and water from urine; storage and distribution of potable water. A simplified block diagram of the water reclamation unit half system is shown in figure 23. Two identical units are provided; each consists of two separate purification paths, one for urine and the other for wash water and condensate.

If electrical power is used for heat at 450 lb/kw penalty, the air evaporation and electrodialysis recovery units for urine appear competitive. However, the utilization of isotope heaters at 100 lb/kw penalty made the air evaporation unit more attractive as shown in figure 24.

The type of urine reclamation unit selected was air evaporation which uses an onboard radio-isotope heat source. A pre-treatment agent is added to the urine prior to introduction into the wick evaporator of the reclamation unit. The unit has a recirculating air stream which evaporates the water from the wicks and carries the water to a condenser. The entrained water is collected and pumped through a millipore filter and by untraviolet lamps.

Wash water and atmospheric condensate is processed by a simpler system which includes a dust filter, mixed ion exchange and charcoal bed, a millipore filter and untraviolet lamps.

In both cases, a bacteria culture test will be made before the water is declared potable. This necessitates holding the water in test tanks for 24 hours prior to transfer to potable tanks. If necessary, the water in the tank may be recirculated through the reclamation unit. A three-day emergency supply of drinking and food reconstitution water will be maintained. Provisions are provided for the circulation of wash water and condensate through the urine reclamation path.

Fecal waste collection takes place in an enclosed, but vented, commode compartment. A blower is utilized to pull cabin air through vents into the commode compartment, and through a permeable plastic bag which passes gas, but not liquids or solids. The airflow helps direct feces into the bag and conducts gases through millipore and activated charcoal filters for odor removal. After defecation, the bag is sealed and placed in a heating unit that dehydrates the feces, drives off gases, and kills bacteria. The gases and distilled fecal water are vented to space. Several commode designs were reviewed and none found acceptable.

Figure 25 indicates the collection and distribution paths of the waste and water management subsystems. As with the atmospheric control and purification subsystem, each of two separate units is capable of supporting the entire 24-man crew.

The urine collection device selected is a centrifugal urinal that uses a small vaneless impeller driven by a spring-wound motor to force the urine through a check valve into the temporary storage tank.

Conclusions and Recommendations

The results of the life support and environmental control system study were used to select subsystems tailored for usage with the LORL configuration.

Study indicates that the LORL configuration permits simplicity in design and considerable time for some equipment "off cycles" and for corrective action in case of failure. Considerable time is also available to take corrective action in the event of a meteorite puncture or to locate an inadvertent leak.

A review of the major subsystems selected for LORL is given below:

Atmospheric Supply and Pressurization

- A Bosch oxygen regeneration unit and subcritical storage of oxygen and nitrogen has been selected for the atmospheric supply system.
- Hangar and airlock decompression is obtained by utilizing a 4 hp compressor/vacuum pump combination and high pressure gas storage in tanks.
- Leak detection is provided by monitoring the use rate of the nitrogen supply and by individual seal sensors.

Atmospheric Conditioning and Purification

- A condenser and wick-type separator and silica gel are used for humidity control, a pair of molecular sieves for CO_2 removal and a catalytic burner for trace contaminant control. Also included are a debris trap, charcoal bed odor remover, chemisorbent bed, fine filter, and ultraviolet lights.
- Twelve space suit stations are provided with umbilicals to accommodate six pressure suits. Each station can operate utilizing the cabin atmospheric control unit or can operate independently.
- Laboratory gas monitoring is provided by a gas chromatograph for periodic sampling of the major atmospheric constituents and a mass spectrometer for identification of trace contaminants.

Thermo-Conditioning

- Space radiators, 1,950 square feet in area, are utilized to reject the laboratory heat load of about 33 kw. FC-75 heat transport fluid is pumped directly through equipment, and cabin heat exchangers for transport of heat to the radiators.
- Isotope heaters are provided for food and water heating, for the reclamation of water from urine and for CO_2 removal unit desorption.

Waste and Water Management

- An air evaporation unit is provided for reclamation of water from urine.
- Wash water is recovered by utilizing mixed ion exchange, filters, and ultraviolet lamps.
- Urine collection is provided by a urinal that uses a small vaneless impeller.
- Each commode compartment has an individual ventilation blower and odor control unit.

The overall weight penalty of the LS/ECS is shown in figure 26. For an initial supply period of 180 days, system weight including fixed weight and consumables is approximately 36,000 pounds. Included in the fixed weight are all LS/ECS equipment, working and reserve fluids (e.g., water, coolant), and the initial atmosphere charge. The consumable weight, approximately 65.5 lb/day, includes food, makeup nitrogen and oxygen, and expendables necessary for water reclamation, atmospheric purification, and waste management.

Acknowledgement

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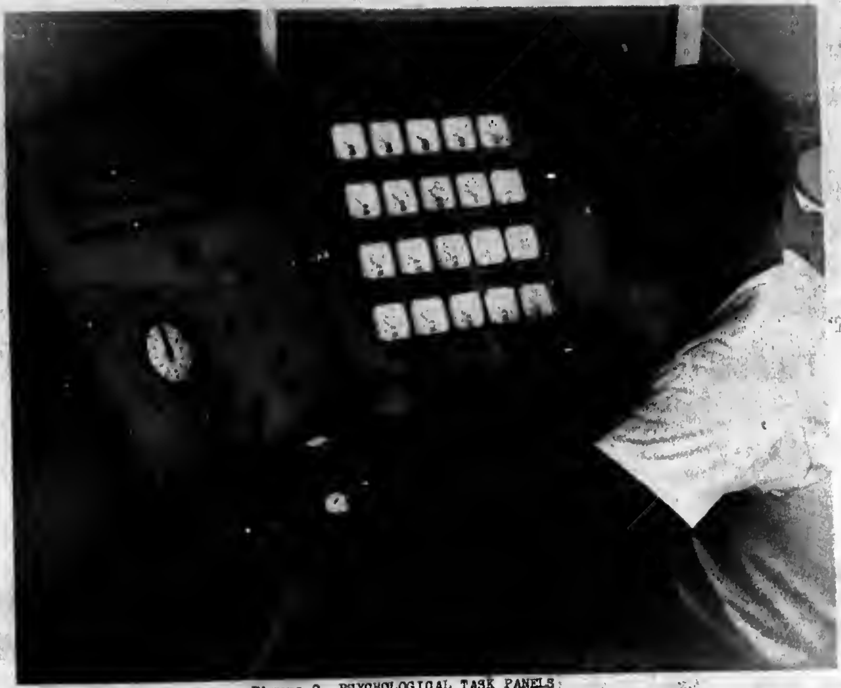

Figure 9 PSYCHOLOGICAL TASK PANELS

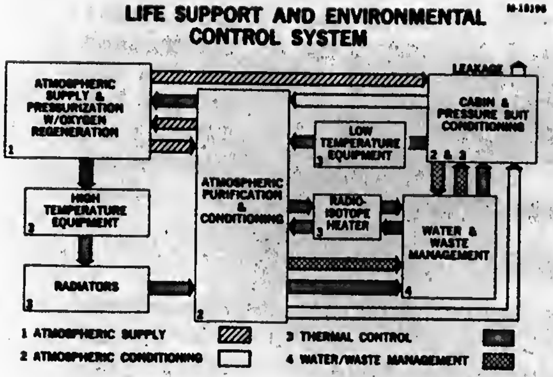
LIFE SUPPORT AND ENVIRONMENTAL CONTROL SYSTEM
N-15196
ATMOSPHERIC SUPPLY & PRESSURIZATION W/OXYGEN REGENERATION
LEAKAGE
CABIN & PRESSURE SUIT CONDITIONING
LOW TEMPERATURE EQUIPMENT
ATMOSPHERIC PURIFICATION & CONDITIONING
HIGH TEMPERATURE EQUIPMENT
RADIO-ISOTOPE HEATER
WATER & WASTE MANAGEMENT
RADIATORS
2 & 3
1 ATMOSPHERIC SUPPLY
2 ATMOSPHERIC CONDITIONING
3 THERMAL CONTROL
4 WATER/WASTE MANAGEMENT

Figure 2

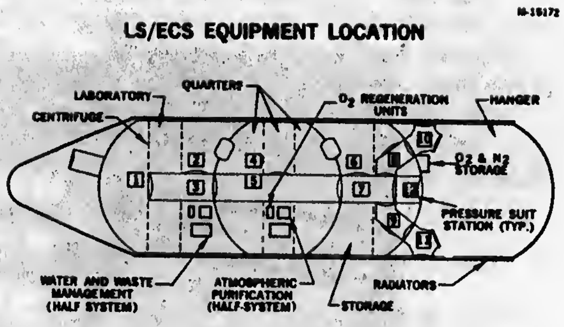
LS/ECS EQUIPMENT LOCATION
N-15172
LABORATORY
QUARTERS
O_2 REGENERATION UNITS
HANGER
CENTRIFUGE
O2 & N2 STORAGE
PRESSURE SUIT STATION (TYP.)
WATER AND WASTE MANAGEMENT (HALF SYSTEM)
ATMOSPHERIC PURIFICATION (HALF-SYSTEM)
STORAGE
RADIATORS
1
2
3
4
5
6
7
8
9
10
11

Figure 3

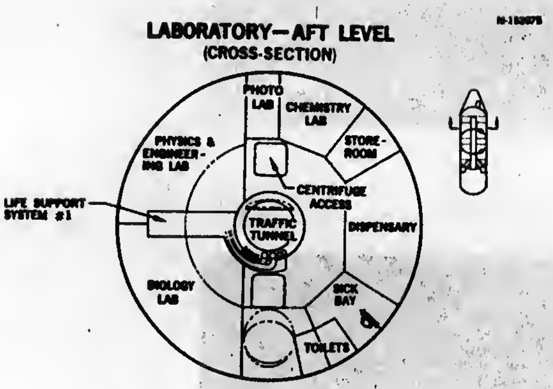

Figure 4

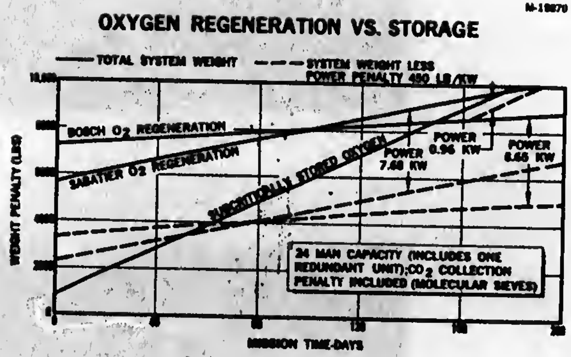

Figure 5

ATMOSPHERIC STORAGE METHODS

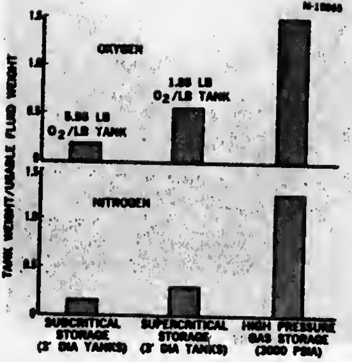

Figure 6

BOSCH OXYGEN REGENERATION UNIT HALF SYSTEM

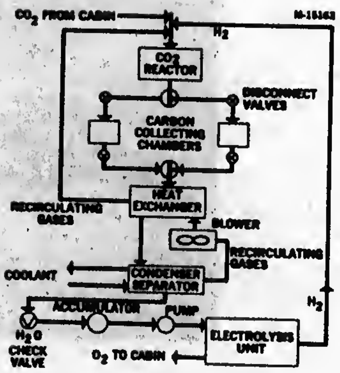

Figure 7

ATMOSPHERIC SUPPLY & PRESSURIZATION SUBSYSTEM

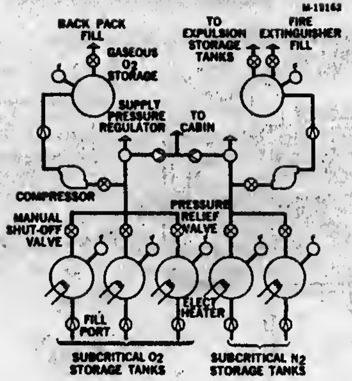

Figure 8

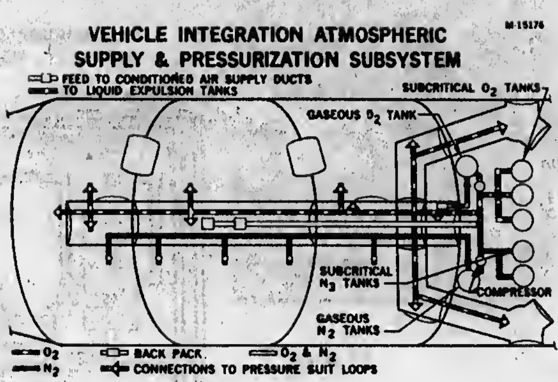
M-15176
VEHICLE INTEGRATION ATMOSPHERIC SUPPLY & PRESSURIZATION SUBSYSTEM
FEED TO CONDITIONED AIR SUPPLY DUCTS
TO LIQUID EXPULSION TANKS
SUBCRITICAL O2 TANKS
GASEOUS O2 TANK
SUBCRITICAL N3 TANKS
COMPRESSOR
GASEOUS N2 TANKS
O2
N2
BACK PACK
O2 & N2
CONNECTIONS TO PRESSURE SUIT LOOPS

Figure 9

HANGAR PUMP DOWN WEIGHT PENALTY

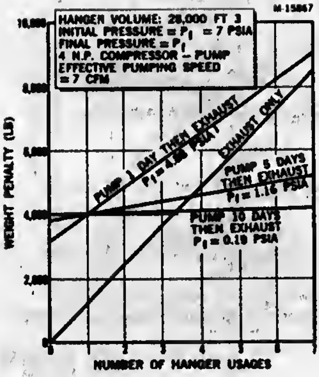

Figure 10

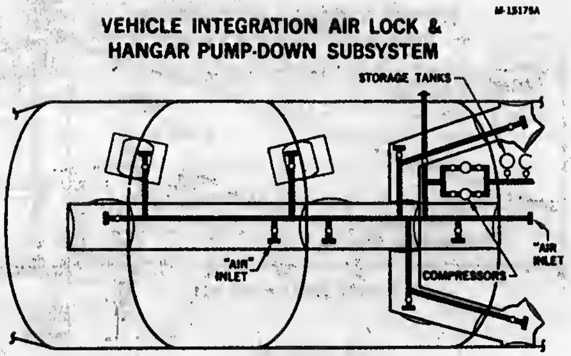

Figure 11

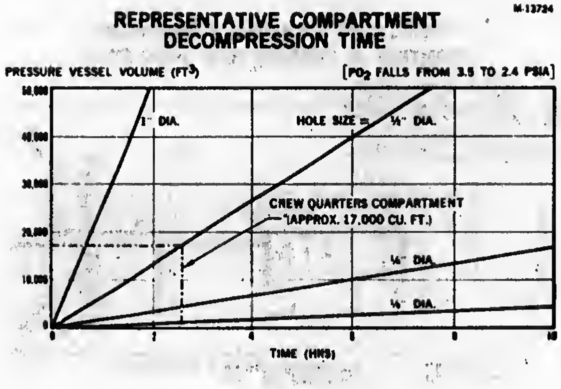

Figure 12

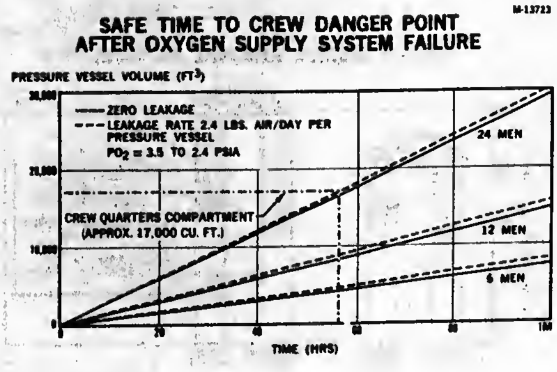

Figure 13

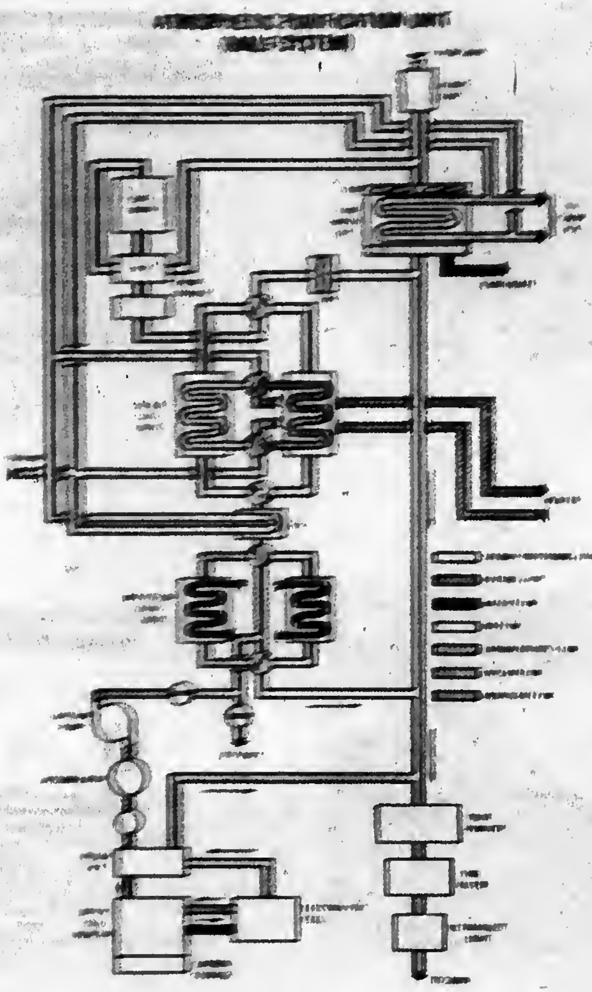

Figure 14

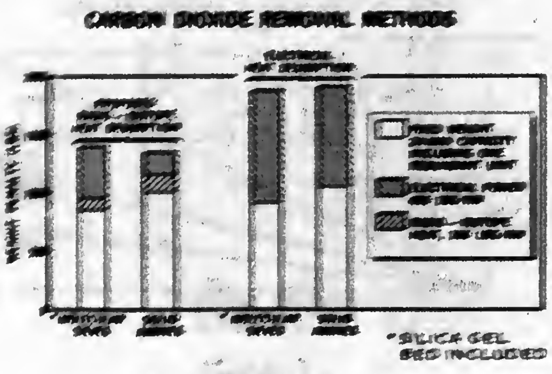

Figure 15

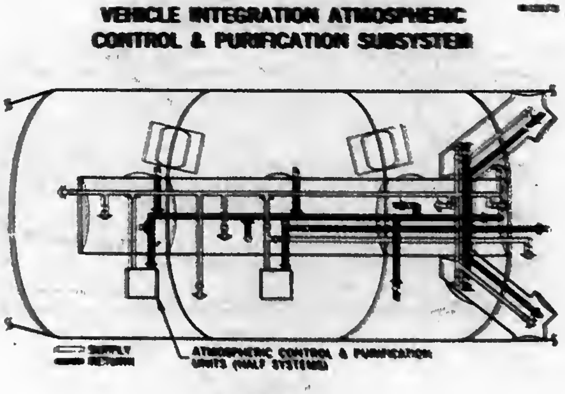
VEHICLE INTEGRATION ATMOSPHERIC CONTROL & PURIFICATION SUBSYSTEM
SUPPLY
RETURN
ATMOSPHERIC CONTROL & PURIFICATION UNITS (HALF SYSTEMS)

Figure 16

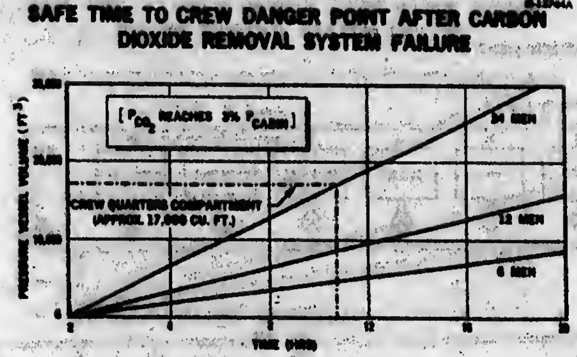

Figure 17

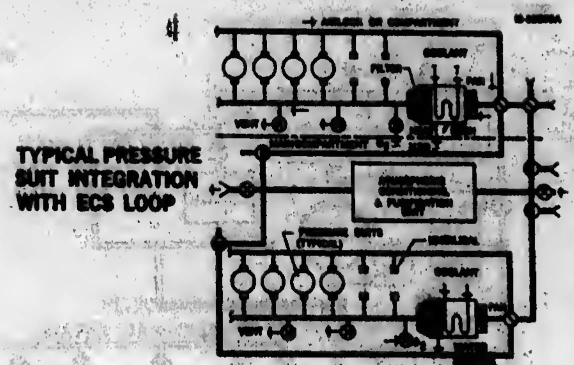

Figure 18

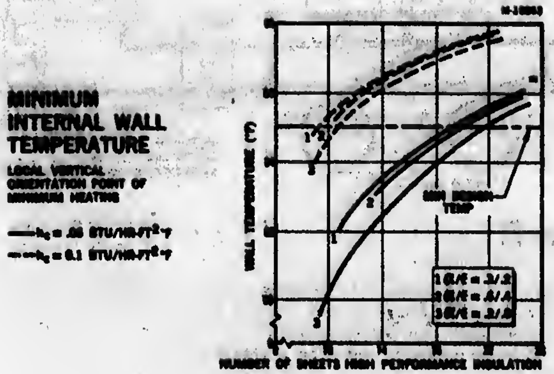

Figure 19

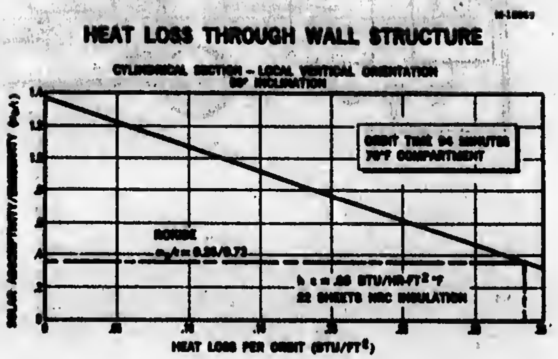

Figure 20

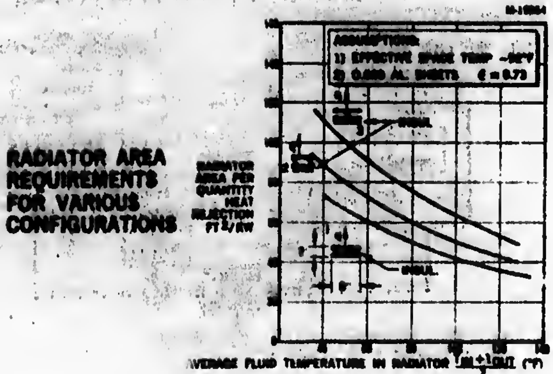

Figure 2:

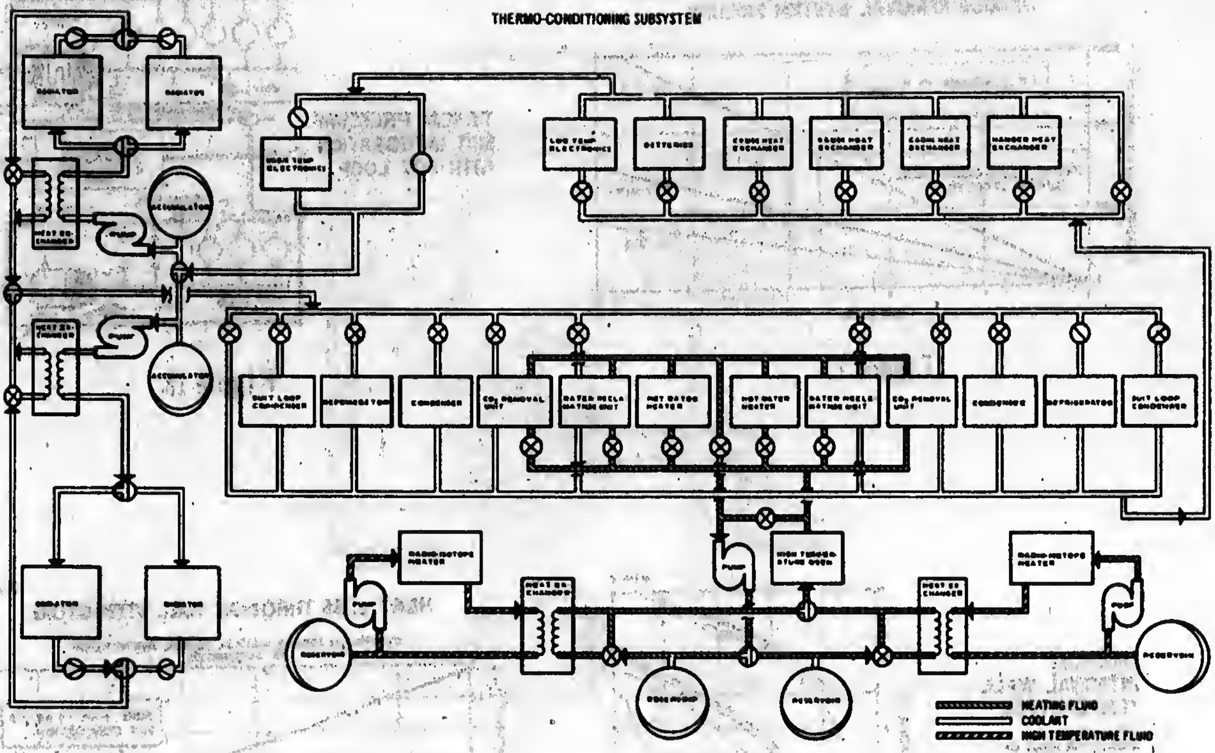

Figure 22

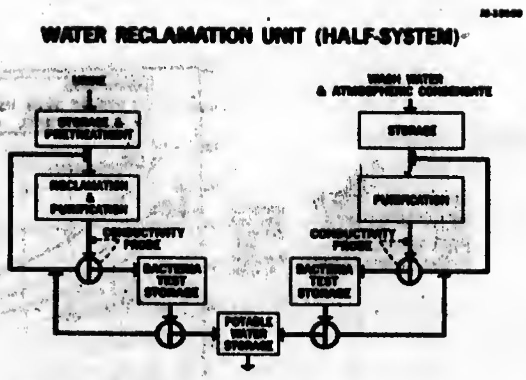

Figure 23

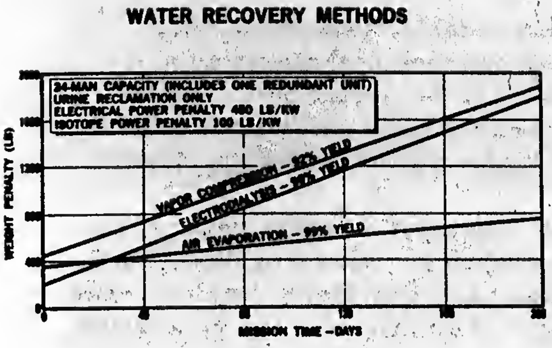

Figure 24

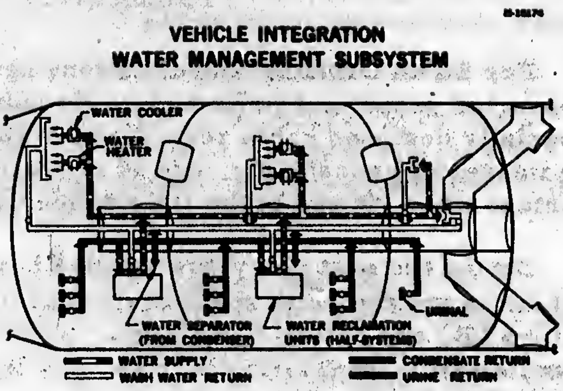

Figure 25

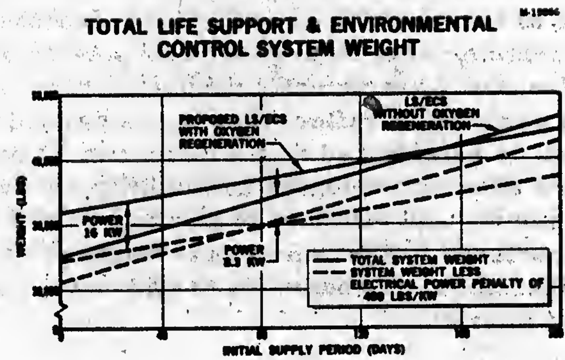

Figure 26

SPACE STATIONS AND SPACE CABIN TESTING

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Carl R. Cording
General Electric Company
Philadelphia, Pennsylvania

For Earth Orbiting Space Stations neither expandable, extensible, nor converted propellant tanks appear as suitable for manned operations as a specially designed cabin regardless of the mission to be performed. While an interesting possibility, use of converted propellant tanks offer little advantage when viewed in the light of the overall space station system problem.

During the past two years studies have been conducted in some depth of the cabins associated with space station systems suitable for launch by Titan III C, Saturn I B, and Saturn 5 boosters. These studies have considered various crew complements, supporting ferries, and the effects of rotation for the generation of artificial "G".

Considering the requirements for integrating power supplies, thermal control, life support, attitude control, orbit propulsion, specific mission equipment, rendezvous, docking, communications, navigation, and crew creature comforts, the development of an efficient usable cabin becomes a task of significant proportions. Applying the constraints of removable and storable equipment to the fixed sizes and shapes of booster tankage makes the problem more difficult, the results less than optimum, and the increased cost substantial.

While cost is always a major consideration in space system design, it is pointed out that space station design reflects major systems costs as a second order function only. The primary influencing factor of space station systems cost is the expense incurred in the associated logistics system. Ferry vehicles and the boosters required to launch them as well as the expense incurred in tracking and recovery operations represent the major portion of total systems cost. Further, since these costs are directly proportional to crew size and tour of duty and since the cabin must reflect these considerations, it is only after a total resource limit is established that a cabin can be designed. At this point, optimization of crew time and volume, availability, reliability, use of existing subsystems equipment, and efficient performance of the primary mission become of major importance. The constraints imposed by the attempted use of existing tankage, therefore, far outweigh the advantage of maximizing the use of structure delivered in orbit.

In addition to booster capability and logistics costs the design of a space station systems is also quite obviously dependent upon the mission to be accomplished. The fundamental unknown to be explored by any early capability space station is man's ability to assist in the optimization of mission performance. Missions can be postulated by the score. They range from those of purely military nature, such as inspection or reconnaissance to scientific evaluation of the environment, astronomy, and others.

Examination of these missions using standard techniques available provide some basic information regarding the function of man in each,and some indication of the cost versus the return of performing these functions in a manned versus an unmanned mode. If all of the postulated crew functions for all of the postulated missions are now grouped by type, a listing as shown on Figure 1 is developed. Here then are the basic modules of crew function which can in the gross sense be selected and arranged for the performance of any military or scientific mission which might be considered.

In the same fashion, operation and maintenance of the station subsystems according to crew function have been grouped in Figure 2. Here the crew's presence has been considered in optimization, repair, selection, and use of the station "machinery" to best carry out the mission of interest.

At this point in the analysis of a manned space system one should be ready to trade-off the benefit of man's performance of the functions above versus the cost incurred by his presence. If this could be accomplished, then the fundamental decision of manned or unmanned system could be made based upon cost, effectiveness, and flexibility. Or if an a priori decision is made that the system is to be manned then the further trade-off of crew size as related to crew functions, degree of automation, crew effectiveness and reliability must be made.

Unfortunately, too little is known about man's performance of these functions in the space station environment to be able to predict with confidence his contribution to the overall system performance. This necessary understanding of man's performance relates not only to his tolerance to weightlessness but also to his ability to remain effective while operating at a high work load within the confines of a small cabin for long durations. Considering the subtle and highly complex nature of the interactions which effect man's performance and the difficulty in developing analytical criteria which is meaningful, the method which appears most suitable for establishing quantitative rationale of the relationship of man to system appears to be through the conduct of a series of high fidelity mission simulation experiments. Intelligently designed, these experiments can provide the basic information required to perform the trade-off studies necessary for space system design.

In March of 1963 General Electric's Missile and Space Division recognized this need for a method of highlighting the most critical factors for early systems design. As a result, a program was initiated for the development of the equipment, the techniques and the trained team which are required to perform laboratory simulation programs of this nature. Seven months later a 30 day closed environment test program was initiated using a four man crew for the prime purpose of evaluating their performance of space station related activities over this long duration.

In developing a program of this nature, it is necessary to review the parameters involved which may have a significant effect upon man's performance. These parameters are shown grouped in three major categories on Figure 3. Although weightlessness and the hazards associated with space flight cannot be simulated, faithful duplication of those remaining variables of total environment, duration, and crew activity can provide sufficient fidelity to the mission situation to permit confident measurement of the crew performance. While these measures will no doubt be somewhat affected by the introduction of zero "g" and fear in actual flight, the simulation conducted permits a much higher degree of confidence and understanding to be applied in relating a crew to a vehicle during the early design phases of system development.

In order to be meaningful, however, a simulation program must be structured as close to the real situation as possible. In all respects this test program used, as criteria and operations specifications, all of the applicable results of the space station study work which General Electric conducted during the past two years. Wherever possible, the equipment and the procedures used followed precisely these systems requirements and design criteria which were developed for an early capability, four to six man, earth orbiting space station.

In design of the test, it was recognized that human performance is always to some extent influenced by surroundings and the environment in which one lives. It was, therefore, necessary to faithfully duplicate a cabin which would be similar in all respects to the best projection of orbital operating hardware which existed. The cabin which was developed for this program is shown on Figure 4. It is 12 1/2 ft. in diameter, 24 ft. high and is separated into living and flight deck compartments.

The living compartment, Figure 5, was designed to convey a feeling of spaciousness and maximize storage volume for necessary equipment and supplies. Pull-out drawers contained food, drugs, and personal equipment for the 30-day mission. A small two cubic foot freezer accommodated diet supplements such as butter, several steaks, frozen apple pies, and tomatoes. A zero "g" type water dispenser was used for reconstituting freeze dried foods.

The center of most activity on the upper deck is the vehicle systems instrument panel shown on Figure 6. This panel is a result of a design study conducted for the development of the display and control requirements of a typical Earth Orbiting Space Station. Specific psychomotor tasks were located adjacent to the main instrument panel.

The cabin was installed as shown on Figure 7 in one of the three 39 ft. diameter space environment simulators now in operation at Valley Forge. Use of this simulator facilitated the contribution of conducting the test in an artificial atmosphere. The atmosphere chosen for evaluation was 7 psia composed of sea level oxygen with nitrogen as the diluent. This choice, the result of a detailed study relating structural penalty, leakage loss, fan power, fire hazard, and extra-vehicular operations appears to be the best compromise between engineering, physiological and operational requirements.

In order to evaluate the physiological effects, if any, of the 7 psia atmosphere selected, an exceptionally complete medical test program was designed and applied to each test subject.

The physiological tests made were separated into two essential categories based upon the equipments and skills required to take the measurements. Those measures requiring the facilities and techniques of medical research installations were generally classified as pre- and post-flight tests and are listed on Figure 8. The pre-flight tests were made to establish baseline data on each subject prior to entry into the cabin. A similar battery of tests was taken immediately upon exit from the cabin and before physiological re-adaptation to the earth-normal environment could have occurred.

The second category of tests included those which could be taken on a daily or periodic basis by the crew during the flight and which required equipment which could readily be expected to be contained within a space station.

These in-flight biomedical measurements were made on each crew member every day and included temperature, blood pressure, electrocardiogram, phonocardiogram and carotid pulse. In addition, pulmonary functions such as vital capacity and timed vital capacity were measured in order to gather the fundamental data required for an evaluation of crew health and well being.

The majority of the activities associated with these measures was contributed by the personnel and facilities of Temple University Medical Center. This joint participation of General Electric's space systems biologists and physiologists with Temple's clinical and research specialists in the areas of pulmonary, renal, and cardiovascular functions permitted an exceptionally extensive biomedical evaluation of the effects of an unusual environment upon man.

Twenty-four hour medical surveillance was also provided in the control room during the flight. Using closed circuit television and the communications system coupled with the daily in-flight data readouts, the doctors on duty were able to closely monitor each subject on a continuous basis. In this way, subjective data of health, well being, morale, and motivation were recorded and correlated with the individual diaries kept by the crew members.

The success of a complex program such as this depends, to a large extent, upon the performance of the team of supporting personnel. This is especially true of those individuals in the role of test monitors who control or have cognizance over all test activities on a minute-by-minute, day-by-day basis. The role of the monitors in this test was maximized in order that a high degree of task and equipment programming flexibility could be realized. For example, the monitors had control over all subsystem conditions, providing the opportunity to judge management performance. Each task was controlled individually, providing programming latitude in case of such things as scheduled emergencies, and equipment failures.

Psychological task panels, shown on Figure 9, were used for evaluation of crew performance. A total of eleven basic psychological tasks designed to probe many aspects of behavior were used for this evaluation. These involved several types of vigilance behavior, eye-hand coordination. higher order mental functions, and reaction time.

Each crew member had a four-hour "mission" period every day, at which time he would perform several of these tasks. Some of the tasks were performed daily, some every other day, and others every third day depending on the particular measure. The tasks were programmed to each individual relative to his work/rest cycle so that factors such as fatigue and alertness were balanced among the crew. This balanced design made it possible to either evaluate the effect of, or eliminate the effect of these factors in analysis of the data. These tasks were used in addition to the more complex operational requirements of rendezvous, docking, and subsystems monitoring.

Using a six degree of freedom real time analog program, the crew was required on a periodic basis to acquire an unmanned supply vehicle at a distance of 20 miles from the station and to fly a completely manual rendezvous mission using the displays shown on Figure 6. The task here was to successfully complete this maneuver and to bring the supply vehicle to a stop within 50 feet of the station with all rates and attitudes at or near zero.

Immediately upon completion of the rendezvous maneuver, the pilot received a visual presentation of the supply vehicle as it would appear to him standing off the station docking port. The presentation was accomplished through the use of closed-circuit television which assumed a camera located at the center of the docking port. The supply vehicle here was the General Electric Docking Simulator, Figure 10, which was operated remotely by the crew member using the same controls he had used to complete the rendezvous maneuver. This docking simulator, an air bearing device in five degrees of freedom, was then maneuvered by the crew-man into the docking port. Of particular interest here is the use of TV visual flight reference only for the performance of this maneuver. It is interesting to note that although the docking maneuver over this two-dimensional presentation system is more difficult than when the simulator is flown with the pilot inside, the maneuver can be accomplished and can be completed in good fashion with a high degree of repeatability.

The Libby, McNeil and Libby Company provided the complete food system for the 30 day test. The diet was primarily composed of freeze-dried food which is reconstituted by adding specific quantities of water, gently kneading the package and in some cases warming for a short time. All of the food provided was analyzed for caloric and mineral content and percentages of carbohydrates, fats and protein. Included in the diet were servings of lobster, crab, roast beef, lamb, a variety of vegetables, deserts and snacks. A number of dishes of this food were prepared for consumption in the zero-g environment by the addition of a sauce containing a colorless and tasteless gelatin, which holds the food together and retains it on a plate or on a fork without the assistance of gravity.

Food preference rankings were made by the crew for each food item of one meal per day for the duration of the test. In this manner a large amount of data concerning the preference of various types of food is available and can be used for structuring the food system chosen for the next test or for a space station program. The importance of food quality to morale was substantiated by the comments of the terranauts.

The crew performed according to the work/rest cycle shown on Figure 11. This schedule was developed during the 30-day test program and is different from the initial cycle developed for the test in terms of the amount of time required for sleep and the balancing of rest periods around the periodic need for food preparation, consumption, and personal hygiene.

During the first several days, the work load required of each man was extremely high with the result that little time was available for eating and personal hygiene, no time was available for rest, and only four to five hours per day remained for sleep. The result -- high exhaustion and loss of morale.

By reducing the data handling work load and allocating more time for sleep, crew effectiveness was restored and morale returned to an acceptable level.

The test was successfully completed on November 6, 1963. Although a wealth of test data exists, discussion of that which appears to be most significant to space station systems analysis design and operation is appropriate.

The work/rest cycle described above has been replotted and is shown on the crew activity apportionment chart, Figure 12. As shown here, each crew-man performed the functions indicated every day during the test. It is significant that of the 24-hour day only four hours per man were available for mission activities. Review of this chart highlights the problem involved in scheduling crew activities and developing a meaningful mission time allocation for crew complements of less than four men.

In the tasks of station management, including all those functions associated with operating the station, the crew performed as well during the last day as they did during the first day. Results of the rendezvous and docking performance indicated an initial difficulty during the first run on the part of each crew member. This difficulty was apparently caused by a loss of proficiency during the three-week period between the end of the rendezvous training and the first simulation in the cabin. This result seems to support the premise that some method of maintaining critical flight phase proficiency will be required on board, and that some time must be allocated for maintenance of this proficiency.

Included in the cabin equipment were several pre-failed electronic modules. Repair of these modules was required on a scheduled basis. In addition, seven unprogrammed failures of operating equipment occurred which demanded repair as soon as possible. In all cases, using schematics and standard checkout equipment the crew was able to trace the failure, effect the appropriate equipment repair, and prevent abort.

Of the eleven specific psychological tests conducted, no decrement of crew performance as a function of time was detected. Conversely, those tests associated with eye-hand coordination indicated a significant improvement in crew performance as a result of the crew's ability to compete one with the other in these particular tasks on a day-to-day basis. Of significance was the high day-to-day variability of the scores recorded during the vigilance tasks. This variability was strikingly large and has been correlated with subjective data gathered through the crew's diaries indicating a strong relationship between morale on any given day with performance of a task which required high concentration over a long period of time. The question one would ask at this time is the effect upon crew reliability during these periods of apparent depression. The answer is yet unknown but certainly indicates that the design of operating equipment must consider the danger associated or implied by these results.

Regarding possible tour-of-duty limitations, the relationship between crew members, as manifested in morale, may be an important factor. For this reason, measures of group cohesiveness were taken seven times during and several times before and after the test. The assessment of cohesiveness was based on a 24-item adjective rating scale which each crew member filled out ranking himself and the three other crew members. Each subject's ratings of psychological distance between himself and the other three crew members was pooled statistically in order to arrive at the measure of cohesiveness shown on Figure 13. As can be seen, the trend over the 30 days is progressively downward. While an acceptable lower limit cannot at this time be established, the possibility of eventual overt conflict is apparent. The rather sharp downward trend in cohesiveness can be partially attributed to the fact that little team performance was required, since most of the mission tasks are, by nature, individual efforts, and therefore, not conducive to development of crew esprit de corps. This explanation is verified by the daily diaries of the crew.

Also, related to tour of duty limitations are the test results associated with nitrogen metabolism. Here, measures of negative nitrogen balance, increased urine, creatinine, blood/urea/nitrogen, coupled with a loss of muscle tone indicates a definite loss of muscle protein and tolerance to exercise. As a result, a daily exercise regimen for future tests, increased substantially for orbital operations while weightless, is indicated. Once again this requirement will have the effect of subtracting from the useful time available for performing mission tasks.

As discussed above, the choice of 7 psia atmosphere (360 mm Hg) was the result of a design study which quantitatively evaluated structural weight penalty, fan power requirements, leakage, purge, and air lock losses. The curve shown on Figure 14 is the result of these trade-offs applied to a four-man space station. As indicated, the 7 psia atmosphere permits a total systems weight reduction of 860 pounds as compared with a sea-level environment. Not only is this weight reduction significant, but in addition the 7 psia environment provides the crew with greater protection from aero-embolism which might possibly be induced by a cabin decompression. It also eliminates the need for the critical time requirement for denitrogenation prior to the use of a pressure suit for extra vehicular operations. These considerations suggest a minimum pressure level of approximately 6 psia. However, the concern for the fire hazard problem associated with high oxygen concentrations is real. The compromise, therefore, between a minimum-weight 6 psia system and one which is reasonable from the fire hazard standpoint appears to be on the order of 7 psia (50% oxygen - 50% nitrogen). While desirable from an engineering standpoint, evaluation of the possible physiological effects upon the crew of long exposure to this environment was necessary.

The measures which were made upon this crew were indicated on Figure 8. The results of these measures shown on Figure 15, grouped by function, indicate completely normal physiological response throughout the 30-day period. All measures taken were in all cases within normal clinically acceptable tolerance limits and could not be considered significant in any respect.

In the operational sense, exposure of test monitors 180 times to the 2.2 critical range of decompression during this test, validated the assumption that problems associated with aero-embolism will not exist if this atmosphere is used. Faulty equipment caused a fire during a pre-test checkout run. This fire was electrical in nature but was easily extinguished using standard fire fighting equipment. This experience also tends to justify the choice, from an operational and safety standpoint of the 7 psia atmosphere.

Of interest to the design of life support equipment are the results associated with the water-balance measures taken. In this particular environment Figure 16 indicates a large increase in sweat and respiration water loss, with a corresponding decrease in urine and fecal water discharge. Of interest also are the critical remarks made by all the crew at a debriefing wherein they indicated a constant feeling of dryness although the humidity level was maintained between 35 and 40 per cent.

The volume of the cabin tested is shown on Figure 17. In all cases the crew reported a complete feeling of adequacy for the duration of the test. Furthermore, it is suspected that this same cabin with only minor modifications could support a six-man crew as well as it housed the four. In terms of arrangement, it was obvious that provisions for individual privacy would be highly desirable and should be considered a design goal. This desire plus the need for increased flexibility of the living compartment indicates a sound and light isolation requirement for the sleeping compartments.

The food which was supplied appear to be an extremely important benefit to the test subjects. Several of the crew indicated that a good meal at the end of their day was anticipated and also looked upon as a reward for their activities and performance. Food preference charts were kept current and are shown on Figure 18. Food preference six is equivalent to "like moderately" while eight is "like extremely". The three day gap shown represents that time when the crew used a pureed food form. Although food preference data was not taken during those three days, the almost unanimous comment was that the food was less desirable than the maximum acceptability diet. In terms of weight, the maximum acceptability diet which was used weighs no more than the squeeze diet sometimes recommended for space flight. The only penalty which is paid, as the lower curves indicate, is the volume required for storage. The trade-off, therefore, becomes a simple one between the volume available and the volume needed to provide this highly desirable food form.

In summary, the results of this test indicate that men can perform adequately for a thirty day mission, that group cohesiveness degraded significantly, that the men showed a definite loss in muscle tone, that 40% relative humidity in the selected atmosphere is too low, that rendezvous and docking can be successfully performed using conventional aircraft-type controls and visual contact, that the effects of the selected atmosphere had no deleterious physiological results, and that freeze-dried foods are highly acceptable and desirable. The cabin arrangement was found generally acceptable. Changes in the work/rest cycle were found necessary after about four days because inadequate time was left for sleeping. Adjustment to schedule that allowed about six hours of sleep daily permitted all tasks to be successfully performed and morale and motivation to rise to a high level for the remainder of the test. Pre-failed panels as well as some unprogrammed failures were successfully repaired with the few tools brought on-board and with a minimum of pre-test instruction as background. At no time was there a threat of abort.

Results also indicated that the four hours per day of mission time per man could be increased by applying more automation to the subsystems monitoring and data handling aspects of the flight. Since any increase in mission time availability is reflected in increased data return, this aspect of station system design and operation is highlighted. This is particularly true if a two-man system is contemplated or if any degree of flexibility to meet unusual or emergency situations is to be realized.

In light of these results, the fundamental assumptions around which this test program was structured appear to be substantiated. First, that a high fidelity mission simulation conducted early in the design phase of manned space vehicles systems is invaluable in providing basic validation of assumptions and development of criteria upon which a system can be analyzed and designed. Second, the use of this type of simulation for the development of specific mission systems operations, crew size and equipment requirements is mandatory for efficient development of specific manned space station systems. Finally, this work has substantiated the introduction of the basic crew utilization considerations such as mission time availability and overall effectiveness as being as fundamental to space station design and as quantifiable as weight, power, volume, and cost.

MANNED MISSION TESTS

	SPACE CABIN TESTS	POSSIBLE FUTURE SPACE CABIN TESTS
STATIC:		
DETECT SURFACE TARGETS	X	X
IDENTIFY SURFACE TARGETS	X	X
INTERROGATE BUOYS		X
TRACK SURFACE TARGETS		X
STATIC AND DYNAMIC:		
DETECT SPACE VEHICLES		
TRACK SPACE VEHICLES		
OPERATE MISSION EQUIPMENT	X	X
SELECT SENSORS, MODES, ETC.		X
ORIENT VEHICLES	X	X
ANALYZE MISSION DATA		X
PROGRAM MISSION		X
TRANSMIT REPORTS		X
DYNAMIC:		
IDENTIFY SPACE VEHICLES		
OPERATE SEARCH EQUIPMENT		
LAUNCH VEHICLES		
RECOVER VEHICLES		

Figure 1 POSSIBLE MANNED MISSION TESTS

SUBSYSTEM TESTS

EQUIPMENT	MAN'S TEST FUNCTIONS	SPACE CABIN TESTS	POSSIBLE FUTURE SPACE CABIN TESTS
STATIC:	ASSEMBLE EQUIPMENT		X
CAMERA RADAR	ALIGN,TUNE EQUIPMENT	X	X
INFRARED TELESCOPE OTHER	OPERATE EQUIPMENT	X	X
	VARY EQUIPMENT PARAMETERS		X
	REPAIR EQUIPMENT	X	X
DYNAMIC:	EVALUATE TEST DATA		X
TRACKING RADAR PROBE VEHICLES	REDIRECT TESTS	X	X
TARGET VEHICLES SPACE PROPULSION OTHER	REPORT RESULTS	X	X

Figure 2 MANNED SUBSYSTEM TESTS

SIMULATION PARAMETERS

- ENVIRONMENT
  - GASEOUS
  - CONFINEMENT
  - CREW EQUIPMENT
- DURATION
  - MISSION DEPENDENT
- CREW ACTIVITY
  - WORK / REST CYCLE
  - MISSION PROFILE AND TASKS
  - MISSION TASKS
  - EMERGENCY CONDITIONS
  - MAN - MACHINE INTERFACE

Figure 3 SIMULATION PARAMETERS

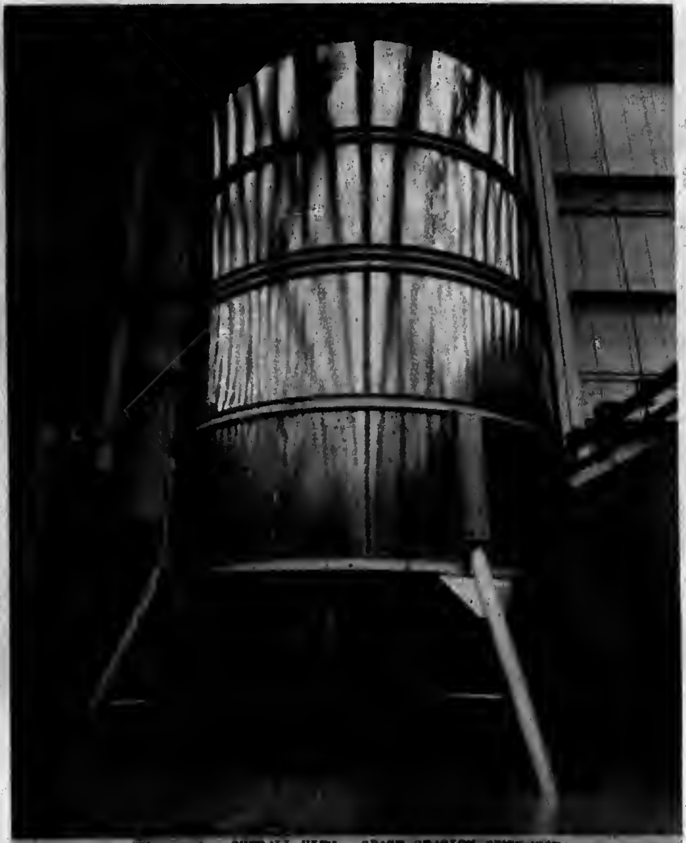

Figure 4 OVERALL VIEW - SPACE STATION SIMULATOR

Figure 5 LIVING COMPARTMENT - FOOD STORAGE AREA

Figure 6 SPACE CABIN INSTRUMENT PANEL

PREFLIGHT/POSTFLIGHT STUDIES

BASIC CLINICAL STUDIES

HEMATOLOGY - Hb, HEMATOCRIT, RbC, WbC, DIFF. PLATELETS, RETICULOCYTES

URINALYSIS - SpG, ALBUMIN, SUGAR, ACETONE MICROSCOPIC

BLOOD CHEMISTRY - BLOOD SUGAR, BUN, CREATININE, Na, K^+, Cl^- CALCIUM, PHOSPHOROUS, PLASMA CATECHOLAMINES

URINARY CHEMISTRY - VOLUME (24 HR), OSMOLARITY, Na^+, K^+, Cl^-, NITROGEN, Ca^{++}, PHOSPHOROUS, CREATININE, CATECHOLAMINES, 17 - HYDROXYSTEROIDS

BACTERIOLOGY - FECAL, PHARYNGEAL, SKIN AND URINARY

ECG

CHEST X - RAY

OPHTHALMOLOGICAL EXAM PERIMETRY

AUDIOMETRY

Figure 8 PRE/POST FLIGHT PHYSIOLOGICAL MEASUREMENTS

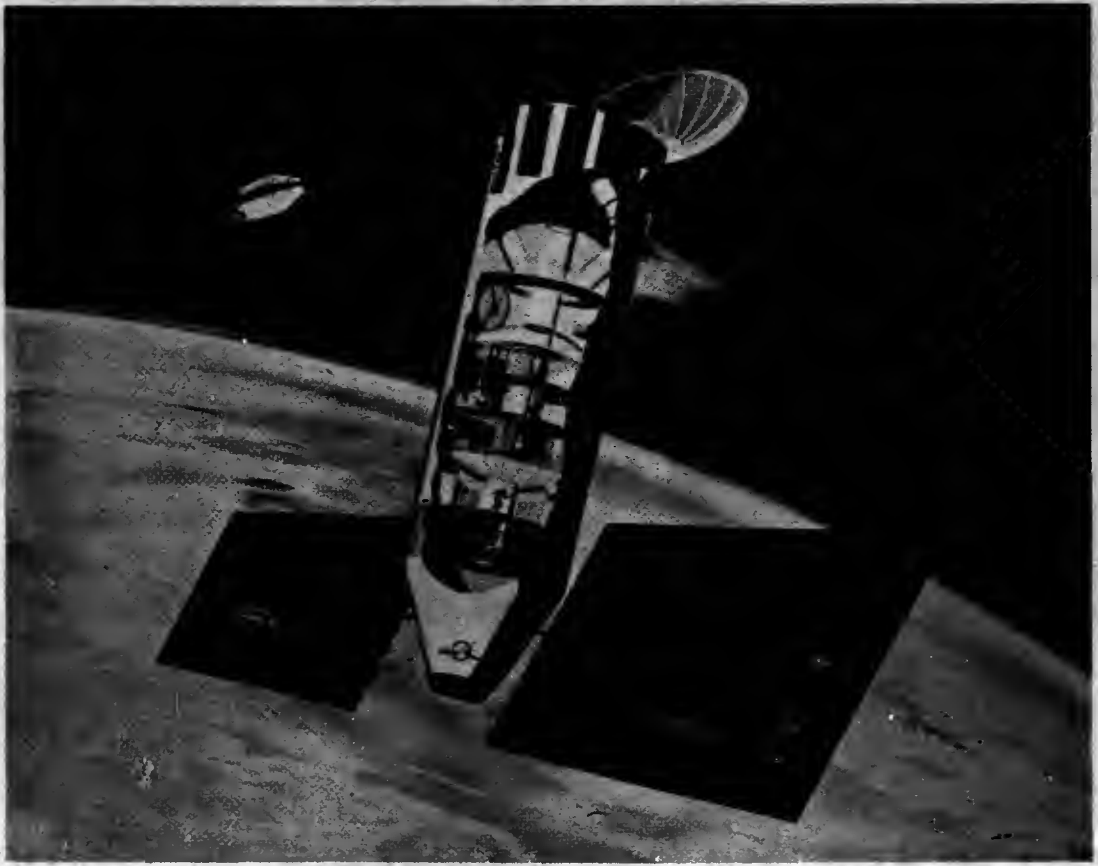

Figure 1. LARGE ORBITING RESEARCH LABORATORY

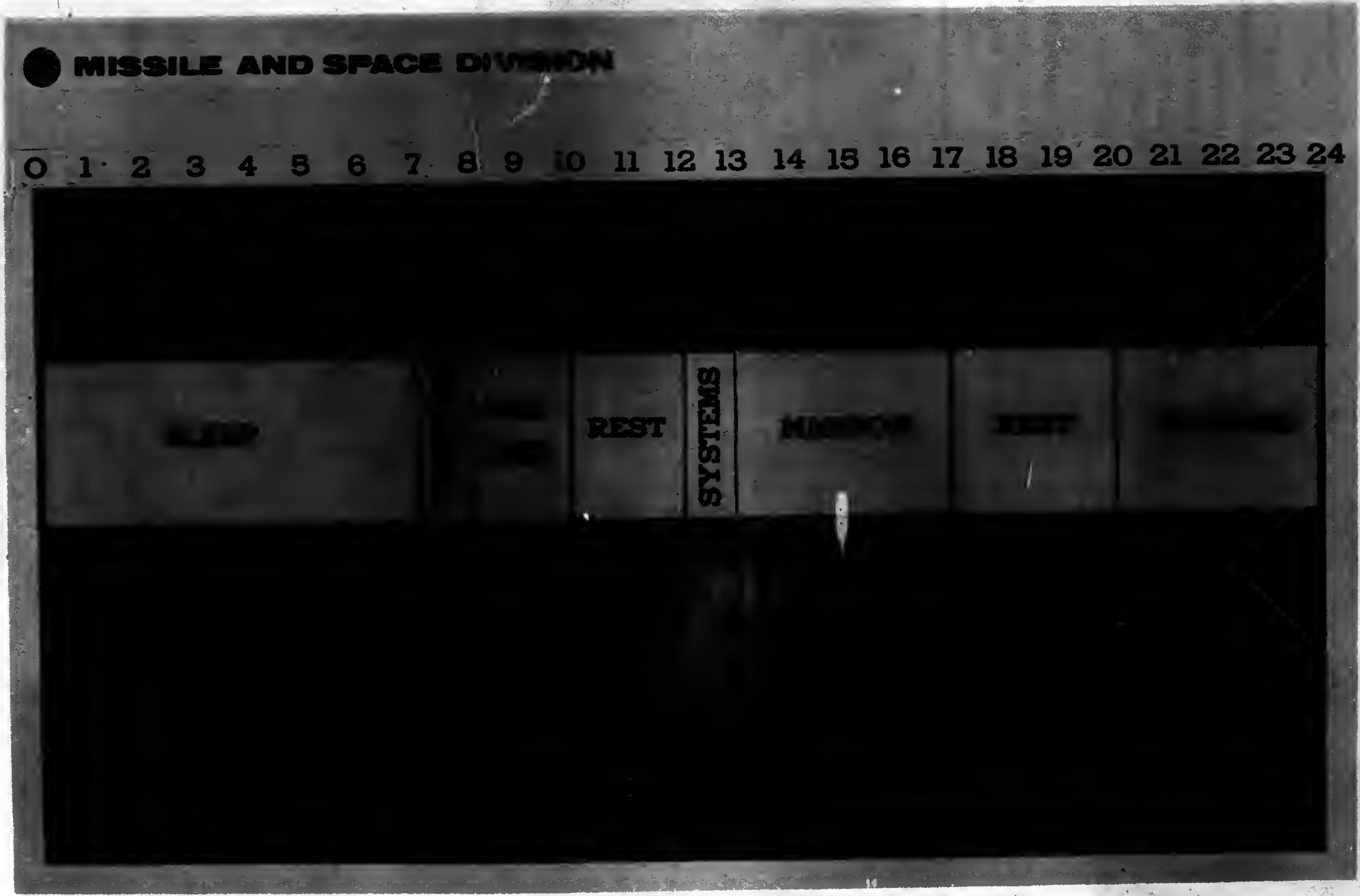

Figure 11 FINAL CREW WORK/REST CYCLE

R. E. JOHNSON

Position: Engineering Specialist
Company: Sylvania Electronic Systems, Williamsville, N.Y.
Education: B.S., Physics, Wisconsin State College
M.S., Physics, University of Wisconsin

Experience: Sylvania Electronic Systems Division Atmospheric Millimeter wave propagation, Electronically Scanned Millimeter Wave Antenna.

Martin-Marietta Corp. - The Martin Company Division, Orlando, Florida, 1958-1963. Senior Engineer to Design Specialist.

Goodyear Aircraft Corp. - Engineer to Senior Engineer, 1955-1958. Research, development, and design of microwave antennas and components.

BIOGRAPHICAL SKETCH NOT AVAILABLE

T. E. WOLK

C. T. MORAVEC

BIOGRAPHICAL SKETCH NOT AVAILABLE

Position Title: Faculty Research Associate
Industrial and Management Engineering Department

Company Affiliation: Newark College of Engineering

Brief Biography: M.S. in Physical Chemistry, Liverpool, England. Post graduate research in electrochemistry in Cambridge and Princeton. Management and research in P. & G., Lever Bros., Educational Testing Service, Parapsychology Foundation, Newark College of Engineering.

E. DOUGLAS DEAN

HUGH LENNEOUS COX

B.C.E., North Carolina State College, 1949; M.S., Civil Engineering, University of Illinois, 1952; Ph.D., Structural Engineering and Applied Mechanics, University of Illinois, 1953.

Joined Martin Company, Denver Division, in 1959. Currently Chief, Structures and Aerophysics Technical Development. Previously served as Initial Head of Technical Development for Silo Launch Test Facility and Technical Director of Complete Titan II Shock Isolation System.

Employed by Kirk Engineering Company (1956-1959) as Consultant to the Martin Company.

Structures and Dynamics Engineer, Army Ballistic Missile Agency, 1955-1956.

Executive Vice President, Structures Specialties Corporation, Santa Monica, California, 1956-1957.

Structural Engineer, Douglas Aircraft Company, Santa Monica, California, 1953-1956.

Numerous honors and awards for technical papers presented.

B.S. in Aeronautical Engineering, University of Kansas, 1949. M.S. in Applied Mechanics, Washington University, St. Louis, Missouri, 1955.

Mr. Winter has been employed since 1957 as a Group Engineer with the Martin Company, Denver. Analysis and test of Titan II Thrust Mount and Shock Isolation System; Titan I Open Base Shock Mounting Inspections; Inflatable Structures Study.

He was previously employed as Stress Engineer, Emerson Electronic Mfg. Company, Electronics & Avionics Division, St. Louis (November 1954 to September 1957).

Mr. Winter taught at Washington University, St. Louis, Missouri, from June 1954 to October 1954.

He was associated with Sverdrup & Parcel Inc., St. Louis, Missouri, as a Designer-Detailer (August 1953 to May 1954).

He is a member of the American Institute of Aeronautics and Astronautics, Tau Beta Pi and Sigma Tau.

RALPH O. WINTER

T. CHARLES HELVEY

Completed doctoral thesis in physical chemistry at Kaiser Wilhelm Institute in Berlin, majoring in chemical engineering. Worked at Medical School of University of Halle (Germany), attended Institute of Nuclear Studies at Oak Ridge. Research associate at Cornell University and University of Miami Marine Laboratory. Associate professor at Oneonta College of New York State University. Joined Research Institute of Advanced Science in Baltimore, subsequently the Human Factors Section of the Martin Company, and headed Environmental and Dynamics Laboratories, Orlando Division.

Visiting professor of biophysics at Radiation Biophysics Department of Kansas University. Director of Biophysics and Astrobiology Branch, Radiation, Inc., Research Division, Orlando; Chief Scientist of Ortronix, Inc., Orlando. In 1960 accepted associate professorship in Biological Sciences at University of South Florida.

Executive Secretary of the Tampa Bay Area Council of Aging, Director of Inter-American Institute for Space Science Education. Author of sixty-two scientific articles and several books.

Mr. Lang received his Bachelor of Mechanical Engineering Degree from the City College of New York in 1955. Upon graduation he joined the Research Division of the Curtiss-Wright Corporation and worked principally on the development of advanced jet engine concepts. In 1959 he became affiliated with the Hamilton Standard Division of United Aircraft Corporation. He was given technical responsibility for the analysis and development of a solar thermoelectric power generator for space use which was designed, buit built and tested successfully within 1 1/2 years. At present Mr. Lang is a System Analyst in the Space Suit Group. His principal responsibility is the performance of optimization studies leading to the development of low weight, small volume, high reliability life support systems. Important among these are the analyses necessary for providing adequate thermal, contaminant, and pressure control. In addition he has coordinated the task of physiological monitoring and evaluation of test data during manned testing.

RONALD LANG

T. C. SECORD

Present Position: Chief, Life and Environmental Systems Section, Advance Space Technology. Directs Advanced Design of Power Systems, Environmental Control and Life Support Systems and Vehicle Safety Systems.

Education: University of Southern California, B.S. in Mechanical Engineering, 1950.

Experience: 13 years at Douglas -- recently directed research, analyses and study programs on MOSS, MORL, OSS, UMPIRE, and S-IV Gemini Space Laboratory, RITA, SLOMAR, ASTRO, Apollo Laboratory and LEM; also worked as Advance Design Supervisor in Mechanical Section, Group Engineer in Air Conditioning Section, and Mechanical Designer in Aircraft Structures Section.

Professional Activities: Director of Southern California ASME Aviation and Space Division, 1960-62; member ASME Professional Division Council, ASME Space Division National Executive Committee (Secretary), ASME National General Committee for Spacecraft, and Institute of Environmental Sciences.

Dr. Yarymovych is Acting Director of Manned Earth Orbital Mission Studies, Advanced Manned Missions Program in the NASA Headquarters Office of Manned Space Flight. He is responsible for the development of the NASA program for advanced manned earth orbital systems, including space stations.

He was previously Assistant Director of Systems Engineering, Flight Systems, in the NASA Office of Manned Space Flight, responsible for systems engineering of various subsystems of the Apollo spacecraft and launch vehicle.

Dr. Yarymovych came to NASA from Research and Advanced Development Division of Avco Corporation, where he was Manager of Nuclear Electric Systems. His industrial experience was preceded by research activities at the Institute of Flight Structures of Columbia University.

He received his B.S. Degree in Aeronautical Engineering at N.Y.U. His M.S. and Ph.D. Degrees in Engineering Mechanics were earned at Columbia University.

MICHAEL I. YARYMOVYCH

RICHARD A. PASSMAN

B.S.E. in aeronautical Engineering and B.S.E. in Mathematics from University of Michigan, 1944. Later, upon discharge from the Navy, was employed as an engineer with Grumman and Consolidated Vultee Aircraft.

In 1947, he received his M.S.E. in Aeronautical Engineering from the University of Michigan.

Was project aerodynamicist at Bell Aircraft Corporation from 1947 to 1956. Projects included the experimental rocket aircraft X-1 and X-2, work on Rascal and Meteor missiles. He joined the General Electric Company in 1956 as project engineer for advanced nose cone systems.

In 1958, was given responsibility for all preliminary re-entry vehicle design. In this capacity the initial designs for Skybolt and Mark 6 were developed. In 1960, he was responsible for Advanced Systems Engineering. In 1961, he was appointed Manager of Advanced Engineering.

His current position is Manager of Advanced Systems Engineering for the Missile and Space Division.

Senior Flight Surgeon in the USAF Specialty Training Program in Aviation Medicine. Assigned for duty with the Deputy for Bioastronautics, AFMTC.

B.S. from Furman University, Greenville, South Carolina, 1949. M.D. from Medical College of South Carolina, 1954. Interned at Methodist Hospital in Gary, Indiana; entered Air Force in 1955.

Completed Primary Course in Aviation Medicine at Randolph AFB, Texas; assigned to Shaw AFB. Was Base Flight Surgeon and Commander of the 363rd TAC Hospital, and later Chief of Professional Services, Office of the Surgeon, Headquarters Ninth Air Force.

Upon completion of training program in radiobiology, was assigned to Office of the Command Surgeon, Air Defense Command, Ent AFB, Colorado, as Chief of Nuclear Medicine.

Received Master of Public Health Degree from Johns Hopkins University in 1962. Assigned to present duty in July 1963.

WILLIAM B. DYE

RAYMOND L. ALLEN

Raymond L. Allen is project engineer on the Dynamic Test Program at Thiokol Chemical Corporation's Wasatch Division, responsible for all interdepartmental technical coordination associated with the installation of Thiokol's new vibration facility and accompanying test program.

He joined the Wasatch Division as an instrumentation and test staff engineer in December 1959. In March 1960, he was promoted to group leader of the Instrumentation and Test Staff. Prior to his present position he served as an assistant project engineer in the Rocket Design Department. Served as a power plant associate engineer at Douglas Aircraft Co., Inc. and as an engineering aide at Aerojet General.

In 1957, Mr. Allen received his B.S. Degree in aeronautical engineering from California State Polytechnic College. Completed a Complex Vibration Seminar in July 1960 at MB Electronics, New Haven, Connecticut; presently studying to earn a master's degree in engineering administration from the University of Utah.

Leonard G. Flippin is associated with the Wasatch Division of Thiokol Chemical Corporation. He has worked with structural dynamics and the mechanics of materials in the Applied Studies Department for three years.

Prior to joining Thiokol, he served four years as a senior structures and dynamics engineer and 13 years as a civil and architectural engineer at Lockheed Aircraft Corporation. Was a senior structures engineer at U.S. Bearing Corporation for one year and a design specialist at Chrysler Corporation's Missile Division for 1 1/2 years.

Mr. Flippin earned his B.S. Degree in Civil Engineering from Lawrence Institute of Technology, Detroit, Michigan in 1949. During World War II he served as a pilot in the United States Air Force. He is a member of Aircraft Owners and Pilots Associates.

LEONARD G. FLIPPIN

A. H. S. CANDLIN

Taught at Northampton College of Advanced Technology (England). Is at Jacksonville University, Jacksonville, Florida, in the Division of Science and Mathematics, and is lecturing to physics seniors on nuclear physics.

Until the beginning of 1960, was on the scientific staff of England's Ministry of Defense and held a series of official appointments within the Defense field, which included wartime service with the Government of India, totalling about fifteen years. At the end of his service in these capacities was engaged part time in space matters.

B.S. in Electrical Engineering, University of Manitoba, Winnipeg, Canada, 1949; Graduate School, Case Institute of Technology, Cleveland, Ohio, 1950. Full M.S. Curriculum less Thesis.

Television Technician Training, Television Training Institute, Philadelphia, Pennsylvania, 1942 (approximately one year). Color Television Training, RCA Institute Home Study Course, 1954.

Radio Operator & Mechanic's School (USAF). Specialist's Training in V.H.F. Receivers and Transmitters (detached service with Royal Air Force).

Employed since 1963 as Department Manager of Quality Systems Engineering, Martin Company, Baltimore. Previous employment with Martin (since 1955) included positions as Quality Manager on Vanguard Program, Titan I Field Crew Effort, Dyna-Soar Booster Program, and Gemini Launch Vehicle Program.

Numerous speaking engagements at Technical Society Meetings, ASQC Conventions, Canadian Aeronautical Institute, etc.

HAGGAI COHEN

L. H. KRATZER

B.S., Mechanical Engineering, University of Utah, 1950. M.S., Physics, University of Utah, 1952.

In charge of the Instrumentation Development Lab, Dugway Proving Ground, 1950-1952. Supervised developmental activities and directed field testing programs on micrometeorological instruments and air sampling devices at Stanford University, 1952-1957.

While a Senior Scientist in R&D, Lockheed Missiles & Space Company, he designed the inertial reference package of the Agena Vehicle. As Research Specialist, he established and directed the operation of a Reliability Diagnostic Laboratory and served as environmental consultant.

In his capacity as Technical Test Director, HIVOS Facility, Mr. Kratzer supervised the activities of up to 65 engineers and technicians.

B.S., Mechanical and Electrical Engineering, 1931, Harbin Polytechnic Institute, Harbin, China; M.S., Mechanical Engineering, 1932, University of Michigan; Ph.D., Mechanical Engineering, 1935, University of Michigan.

Designer and Draftsman, Economy Baler Company, Ann Arbor, Michigan, 1935-1936. Instructor, Fenn College, Cleveland, Ohio, 1936-1940. Acting Head, Mechanical Engineering Department, Fenn College, 1940-1944. Project Engineer, Thompson Aircraft Products, Cleveland, Ohio, 1944-1945. Acting Head, Fenn College Mechanical Engineering Department, 1945-1946. Senior Industrial Engineer, Kenneth A. McIntyre Associates, Cleveland, Ohio, 1946-1947. Aeronautical Research Scientist, NACA, Lewis Laboratory, Cleveland, Ohio, 1947-1953. Design Specialist, Hydro-Aire, Inc., Burbank, California, 1953-1954. Engineering Specialist, Northrop Aircraft, Inc., Hawthorne, California, 1954-1957.

Joined Lockheed Missiles & Space Company in 1957. Currently Senior Staff Engineer, serving as consultant to laboratory engineers.

(Picture Not Available

VASILY D. PRIAN

Graduated from M.I.T. in 1959 with a B.S. in Optical Physics. In January of 1958, he started half-time employment with Block Associates, Inc., of Cambridge, where he worked on a metrological interferometer and design and fabrication of infrared spectro-radiometers. Also during his years at M.I.T., he was responsible for the operation and maintenance of the Color Measurements Laboratory of M.I.T.

Joined the Research Laboratories of United Aircraft Corp. in February of 1960, where he developed automated lens design and evaluation programs using 7090 computers.

Mr. Willey went into full time activity in his own company, Willey Optical Research Lab and Development Service (WORLDS, Unlimeted), in July of 1962. WORLDS was engaged in engineering, lens design, and prototype fabrication.

In July of 1963, WORLDS was purchased by the Instrument Corporation of Florida and Mr. Willey assumed the directorship of their combined optical activities.

RONALD R. WILLEY, JR.

HAROLD L. JURY

Presently the Project Manager of Advanced Geodetic Survey Projects for Pan American World Airways at the Atlantic Missile Range, Mr. Jury holds a B.S. Degree in Geology from Syracuse University and a M.S. Degree in Photogrammetry and Geodesy from Ohio State University. While completing graduate studies at Ohio State, he served as a research fellow doing photogrammetric research in Greenland.

Mr. Jury worked three years with the Inter-American Geodetic Survey in Central and South America, performing various types of surveys and training Latin engineers in the sciences of geodesy and photogrammetry.

During World War II, he was a combat navigator on B-29 aircraft in the Pacific and presently holds a reserve commission in the United States Air Force. He is a member of the American Society of Photogrammetry and the American Institute of Aeronautics and Astronautics.

Dr. Siegmund was born in Germany but emigrated to the United States in 1930. He grew up in Rochester, New York and pursued his undergraduate and graduate studies at the University of Rochester, where he received his Ph.D. degree in 1952.

After a year of post-doctoral work at Rochester he joined American Optical Company in 1953 as a member of its Research Department staff. He became Assistant Director of Research in 1958, a position he held until December 1963, when he became Manager of the Fiber Optics Department of the Space Defense Division.

(Picture Not Available)

WALTER P. SIEGMUND

THOMAS J. HAYES III

Thomas J. Hayes III, son of Major General and Mrs. Thomas J. Hayes (USA-ret), was born 26 August 1914 in Omaha, Nebraska. A graduate of the U.S. Military Academy at West Point (Class of '36) with a Master of Science Degree from MIT (1939). Also a graduate of The Engineer School, the Command and General Staff College, and the Industrial College of the Armed Forces.

His 27-year career in the Army Engineers has been a varied one. In addition to troop duty with Engineer units at home and abroad, he has served on the faculty of The Engineer School, Fort Belvoir, as Engineer Liaison Officer to the British Army, Assistant Military Attache in London, and Assistant Engineer Commissioner of the District of Columbia.

General Hayes was in charge of the $1.7 billion construction program developing the nationwide network of intercontinental ballistic missile bases for the Air Force, and two years ago was selected to head the Corps of Engineers' activities supporting the NASA Manned Space Program.

Born in Scranton, Pennsylvania, August 18, 1911. Graduated from Scranton Technical High School in 1932; received B.S. in Economics from the University of Notre Dame in 1936. Entered New York University Law School, receiving his L.L.B. in 1940.

After practicing law in Elizabeth, New Jersey, was counsel for the U.S. House Committee on Administrative Law. After the War, served as counsel to the U.S. Senate Small Business Committee, and later as a Production Analyst for the U.S. Navy in Washington.

Joined Bogue Electric, Inc., Washington, D.C., as Vice President in charge of production. In 1955, became Senior Management Consultant for Wellings-Reed, Inc., Washington, D.C. Remained with this firm until 1960; accepted a position with the Federal Aviation Agency, continuing with this agency as Management Analyst Supervisor until 1962. Currently Chief, Management Analysis Office, Executive Staff, of the George C. Marshall Space Flight Center, Huntsville, Alabama.

(Picture Not Available)

JOSEPH H. REED

(Picture Not Available)

BEN W. BRION

Ben W. Brion was graduated with a Bachelor of Science Degree in Electrical Engineering, major in Communications, from Purdue University in 1937.

Prior to World War II he worked as a field engineer testing gun control prototypes for Sperry Gyroscope. During the war he moved to Minneapolis Honeywell where he designed military and commercial control systems.

In 1947, with the Engineering Research Division of Remington Rand, he served as Project Engineer on that company's first electronic digital computer. From Remington Rand he moved to the Mechanical Division of General Mills, where he held the position of Chief Electrical Engineer for eight years.

In 1958 he started his own company, the Brion Engineering Company, where he engineered and marketed two products.

Brion is presently a Senior Staff Engineer with General Electric's Apollo Support Department in Daytona Beach, Florida.

Employed by the Martin Company Canaveral Division as a mechanical systems engineer assigned to the Titan III Project. Formerly Senior Staff Engineer in the Ferrite Components Section of Sperry Microwave Electronics Company in Clearwater, Florida.

Was employed as a Registered Professional Engineer in Chicago until moving to Florida in 1957. Educated at Illinois Institute of Technology.

C. D. SCHWEBEL

Frank B. Page, Ph.B., Yale University, 1931, taught engineering at the University of Rochester, and was an engineer at Pratt & Whitney Aircraft, at Brookhaven National Laboratories, and at General Electric.

While teaching, he served as consultant to General Motors on aircraft powerplant controls. He joined G.E. in 1953 as an engineer in the General Engineering Laboratory, where he did development work in pumping and propulsion systems; served the Small Aircraft Engine Department as a reliability engineer.

In 1962 he joined the Apollo Support Department, and was assigned to Huntsville, where he laid the groundwork for current G.E. reliability support at Marshall Space Flight Center. He is co-holder of two patents, and author of a number of G.E. technical reports.

He is a member of ASME, ASEE, and a charter member of the Daytona Metropolitan Section of AIAA.

(Picture Not Available)

FRANK B. PAGE

George E. Henry, A.B., University of North Carolina, 1945, has been with General Electric since 1948, first as a member of the General Engineering Laboratory in Schenectady, later as manager of Reliability for Ordnance Department in Pittsfield.

He joined the Apollo Support Department in 1962, and has done development work in acoustics and electrical controls. Holds a number of patents, has been active as a lecturer and teacher of Company-sponsored courses. Has written for G.E. Review, Scientific American, and the new Grolier Encylopedia. Is a charter member of the Daytona Metropolitan Section of AIAA.

GEORGE E. HENRY

Mr. Zachmann has more than 20 years experience in electric motors and power systems. Since joining Martin in 1951, he has served as consultant on electric motor applications, power systems, batteries and electromechanical actuators. Most recently, he has been active in the design of electrical power systems for space and lunar applications.

Prior to joining Martin Marietta, he has worked in a wide variety of industries. He has acquired a depth of experience in research, development, design and application of electric power equipment including rotating machines, turbo-alternators, transformers, switch gear, batteries and electromechanical devices. His numerous papers qualify him as an authority on aerospace electrical systems.

A native of New Jersey, he holds a B.S. in Electrical Engineering from Newark College of Engineering and is a registered Professional Engineer in the states of New York and Pennsylvania.

(Picture Not Available)

HOWARD G. ZACHMANN

DARREL G. HARDEN

Assistant Professor of Mechanical Engineering at the University of Oklahoma, Norman, Oklahoma. Teaches and researches in the heat transfer-fluid flow-thermodynamics area.

Head, Department of Mechanical Engineering, Oklahoma State University, Stillwater, Oklahoma. Administers large graduate and undergraduate program at Oklahoma State. Active in all phases of heat transfer research.

(Picture Not Available)

J. H. BOGGS

DAVID K. BARTON

David K. Barton was born in Greenwich, Connecticut, in 1927. He received the B.A. Degree in Physics from Harvard College in 1949.

From 1949 to 1953 he worked as an electronic engineer with the White Sands Signal Corps Agency. From 1953 to 1955 he was a project engineer in Evans Signal Laboratory, Belmar, N.J., responsible for development contracts on radar beacons and related equipment. Between 1955 and 1963 he was a systems engineer with RCA Missile and Surface Radar Department in Moorestown, New Jersey.

In 1958 he received the David W. Sarnoff Award for outstanding achievement in engineering, based upon his contributions to precision tracking radar. In 1963 he joined the Raytheon Company at Wayland, Mass., as a staff engineer. Has presented papers at national conventions and symposia, was a lecturer at the 1960 and 1961 Special Summer Course in Modern Radar Technique at the University of Pennsylvania.

Edward Heinzerling received his B.S. Degree in Electrical Engineering from Tufts College in 1951. From 1951 through 1954 he served as an Electronics Officer in the U.S. Navy.

In 1956 he received the M.S. and E.E. Degrees in Electrical Engineering from the Massachusetts Institute of Technology. Since 1956 he has been with the General Electric Company, Syracuse, New York, where he specialized in the analysis and evaluation of missile radio guidance and tracking equipment.

EDWARD HEINZERLING

J. N. STRAND

Mr. Strand is employed by General Electric in their Radio Guidance Operation. He has been employed by them for the past three years as a Mod III Radar Systems Evaluation Engineer and has completed the National Bureau of Standards 1962 course in Radio Propagation.

For the past two years he has been evaluating the effects of the troposphere upon noise in radar data from Mercury, Ranger and Mariner missions. Previously, he was employed by Boeing Airplane Company as a test planner for the radio controlled Bomarc. Mr. Strand graduated from Illinois Institute of Technology in 1956 with a B.S. in Chemical Engineering. He is a member of the American Ordnance Association.

Personnel Psychologist with fourteen years experience in industry. Completed M.A. in Psychology at Fordham University in 1951 and Ph.D. in Industrial Psychology at Western Reserve University in 1956.

Work centered on executive evaluation and management development. Having formerly worked in the consulting field, now serving as Director, Personnel Planning, Development, and Training with ITT Federal Laboratories.

Member of American Psychological Association and author of several articles in professional and management periodicals.

ARTHUR D. KELLNER

Mr. Lazar is a Personnel Development Specialist with ITT Federal Laboratories, a multi-plant electronics R&D operation of 5,000 people. In this capacity he is responsible for management development activities, which include performance appraisal and individual development, maintenance and operation of the company skills inventory, test evaluation, personnel research and control of turnover. Prior to his present position, Mr. Lazar worked for 5 1/2 years in Human Engineering.

He received his B.B.A. Degree in Industrial Psychology in 1956 from CCNY and his M.A. Degree in Psychology from Syracuse University in 1957. He is currently working for a Ph.D. at NYU.

RICHARD G. LAZAR

Laurence W. Enderson, Jr. is an Aerospace Engineer in the Mission Analysis Section of the NASA Langley Research Center. He is currently doing research in Space Mechanics on problems associated with manned space exploration. Prior to this he was employed by the NASA Manned Spacecraft Center in the Mercury Project Office.

(Picture Not Available)

LAURENCE W. ENDERSON, JR.

(Picture Not Available)

William H. Michael, Jr. is head of the Mission Analysis Section at the NASA Langley Research Center. His research interests have included trajectory analysis for manned and unmanned lunar missions, feasibility studies for experiments in space exploration, and studies in celestial mechanics.

WILLIAM H. MICHAEL, JR.

Gordon D. Thayer was born October 24, 1931, at Glen Ridge, New Jersey. He attended Cornell University (1949-1951) as a math major; Newark College of Engineering (1954-1955), M.E. major; and the University of Colorado (1955-1957), receiving a B.S. Degree in Engineering Physics June 9, 1957.

Joined Boulder Laboratories, National Bureau of Standards, June 17, 1957. Presently with the radio-meteorology group, his projects with NBS include tropospheric analysis, mathematical analysis and data reduction, research on problems of tropospheric refraction and attenuation of radio signals.

Prior to joining NBS, Thayer was employed as a Chemistry Laboratory Aide at the Cathode Ray Tube Division of Allen B. Dumont Labs., Allwood, New Jersey, and as Instrument Calibrator and Checker at Weston Elec. Instr. Corp., Newark, New Jersey. While in the U.S. Army Signal Corps, he was at White Sands Proving Ground and at Evans Signal Lab, Belmar, New Jersey.

(Picture Not Available)

GORDON D. THAYER

ARTHUR A. DAUSCH, JR.

Born in Chicago, Illinois, 1920. B.S. in Electrical Engineering, University of Southern California, 1944; M.S. in Electrical Engineering, University of Southern California, 1950.

Was with Lockheed Aircraft Company in their Experimental Flight Division while earning his B.S. Degree. Commissioned as Ensign, USNR, in August, 1944, and assigned to the Aircraft Electrical Division of the Naval Research Laboratory in Washington, D.C. Returned to the University of Southern California in July, 1950, where he was Head of Evaluation for the Guidance and Control Department until December, 1954. Joined Lockheed Missile System Division in Van Nuys, California.

In 1958 he returned to Hughes Aircraft Company, in the MG Series Fire Control Systems Department. Joined the Reliability and Systems Test Department of the Surveyor Spacecraft Laboratory in 1960; currently Assistant Manager of the Surveyor Systems Engineering Department.